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Space Programs Summary No. 37-37 Volume I

for the period November 1, 1965 to December 31, 1965

The Lunar Program

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I. Surveyor Project

A. Introduction

Calculated to span the gap between the *Ranger* flights and Project *Apollo*, the *Surveyor* spacecraft is designed to take the next step in lunar technology by attempting soft landings on the Moon. The first launches will be engineering test missions to demonstrate system capability up to soft landing, and limited post-landing operations. The engineering payload includes elements of redundancy, diagnostic telemetry, touchdown instrumentation, and survey TV.

Following the engineering test missions, *Surveyor's* objectives are to extend our knowledge of lunar conditions and to verify suitability of *Apollo* landing sites.

Hughes Aircraft Company, Space Systems Division, is under contract to develop and fabricate the first seven spacecraft. The launch vehicle is a combination *Atlas/Centaur*. The JPL Space Flight Operations and Deep Space Network (Mission Operations System) will be utilized for flight control and tracking. The first launch is presently anticipated for the second quarter of 1966.

The major portion of the following material relating to the *Surveyor* spacecraft system was derived from material

prepared by the Hughes Aircraft Company; articles prepared by the JPL Technical Sections are identified by footnotes to the article titles. The principal model designations and vehicles discussed are defined in Table 1.

Table 1. Model and vehicle designations

Designation	Description
SC-1-7	Flight spacecraft with engineering payload
SC-8-10	Flight spacecraft with scientific payload
A-21	Model designation for spacecraft with engineering payload (SC-1-7)
A-24	Model designation for spacecraft with scientific payload (SC-8-10)
T-21	Model A-21 prototype system test spacecraft
T-2	Simplified spacecraft for evaluation of terminal descent propulsion and flight control
S-2	Test spaceframe for vibration, shock, and static structural tests
S-6, -7	Test spaceframe for vernier propulsion system tests
S-9	Structural test model
S-10	Temperature control model

B. System Engineering

1. A-21 Spacecraft Configuration

Model A-21 spacecraft carry an engineering instrument payload to evaluate spacecraft performance and lunar soft landing characteristics. In December 1965 it was directed that A-21 model spacecraft will be used for the first seven, instead of the first four, missions. The current A-21 weight summary is shown in Table 2. The A-21 balance summary showing the cg location is given in Table 3.

a. SC-1 preliminary consent-to-ship review. The consent-to-ship review is a major milestone in a series of reviews that has taken place during the development program on SC-1. Preliminary SC-1 consent-to-ship meetings were held on October 6, 7, 8, 20, and 21 during which problem lists was discussed. The material in the consent-to-ship problem lists was prepared to assist the combined systems test review board in determining whether the SC-1 *Surveyor* spacecraft is in an acceptable state of readiness when shipped from the Hughes Aircraft Company facility at El Segundo, California to the launch facility at the Air Force Eastern Test Range (AFETR).

On July 1, 1965 a quality assurance system verification review was held to assess the effectiveness of this activity

on the *Surveyor* program. The output of this review was a list of open items considered necessary to bring the quality assurance system up to a satisfactory level. On July 7, 1965 an SC-1 design verification review was held to verify the adequacy of the SC-1 design. The output of this review was a list of SC-1 open items requiring additional action prior to launch. In September 1965 a series of unit consent-to-ship meetings was held to review in detail the status of each hardware item at the unit level. As each major test activity on SC-1 was initiated, a consent-to-test meeting was held and open items were generated and reviewed.

Open items from these reviews, as well as items from other sources have been collected into the spacecraft subsystem groupings and have been analyzed as to the effect on the mission. These open items and the recommended disposition are presented in the consent-to-ship problem list document by subsystem grouping. These open items include: (1) quality assurance discrepancy list, (2) design verification review, (3) JPL action directives, (4) engineering change actions, (5) trouble and failure reports, and (6) unit consent-to-ship reports. In addition to the open items, this document also contains spacecraft configuration data, and reliability estimates based on SC-1 system test data and special test vehicle data. Finally, the elements and procedures necessary to ship, receive, handle, and operate the spacecraft were reviewed. This includes the readiness of ground equipment, AFETR schedules, facilities, staff procedures, packaging, shipping, and safety.

The problems were dispositioned by the consent-to-ship review board into four categories:

- (1) Items that have an insignificant effect on the probability of meeting the Mission A objectives and need not be corrected prior to launch.
- (2) Items that have a significant effect on the probability of meeting the Mission A objectives and must be worked prior to launch of the spacecraft.
- (3) Items that have a significant effect on the probability of meeting the Mission A objectives and must be worked prior to shipment of the spacecraft.
- (4) Items that have insufficient data at the present time to determine the probability of meeting Mission A objectives and therefore require additional staff work for evaluation and review.

The items in Categories 2, 3, and 4 are listed in SC-1 consent-to-ship open item summaries and are included in the open item summary check list. Items in these categories

Table 2. A-21 spacecraft weight summary

Item	Current design weight, lb
Total separated weight	2193.52
Basic bus	707.65
Usable propellant	1422.37
Engineering payload	63.50

Table 3. A-21 spacecraft weight and balance summary

Condition	Weight, lb	Vertical center of gravity z ^a , in.	Lateral center of gravity x, in.	Lateral center of gravity y, in.
Weight after Centaur separation	2193.52	11.96	0.05	-0.20
Maximum touchdown weight	645.58	16.86	0.03	-0.19
Minimum touchdown weight	619.35	17.65	0.01	-0.23
^a Inches above landing gear hinge line (spacecraft Station 47.48).				

will be cleared before the final consent-to-ship meeting. The items dispositioned by the consent-to-ship board as Category A are "acceptable for flight" and are deleted from all subsequent open item summaries for SC-1.

b. System and subsystem reliability estimates. The current *Surveyor a priori* reliability estimates by subsystem are shown in Table 4. For the flight and landing mission, the *a priori* reliability of the A-21 spacecraft is 0.812. The reliability estimate based on test data for the SC-1 spacecraft for the flight and landing mission is 0.584. SC-1 reliability estimates by subsystem are shown in Table 5.

Table 4. A-21 subsystem *a priori* reliability (flight and landing mission)

Subsystem	Reliability
Telecommunications	0.975
Vehicle mechanism	0.986
Propulsion	0.922 ^a
Electrical power	0.959
Flight controls	0.955
System	0.812
^a Excludes vernier engine thrust chamber assemblies.	

Table 5. SC-1 subsystem reliability estimates based on test data (flight and landing mission)

Subsystem	Reliability
Telecommunications	0.949
Vehicle mechanisms	0.854
Propulsion	0.924 ^a
Electrical power	0.926
Flight controls	0.843
System	0.584
^a Excludes vernier engine thrust chamber assemblies.	

Table 6. A-21A spacecraft weight summary

Item	Current design weight, lb
Total separated weight	2533.67
Basic bus	743.43
Usable propellant	1657.20
Scientific payload	133.04

Table 7. A-21A spacecraft weight and balance summary

Condition	Weight, lb	Vertical center of gravity z ^a , in.	Lateral center of gravity x, in.	Lateral center of gravity y, in.
Current design weight				
Weight after Centaur separation	2533.67	11.57	-0.16	-0.00
^a Inches above landing gear hinge line (spacecraft Station 47.48).				

2. A-21A Spacecraft Configuration

The current A-21A weight summary is shown in Table 6. The A-21A balance summary showing the center of gravity location is given in Table 7.

C. System Analysis

1. Surveyor Mission A Trajectory Design

Current *Surveyor* Mission A trajectory design studies have been made to include the direct ascent launch periods of April, May, and May/June 1966. The mission design constraints, ground rules, and input data used in trajectory design are given in Refs. 1-5. Representative data are taken from the May to June launch period.

Fig. 1 shows the launch windows for all days of the launch period. Figs. 2 and 3 show unbraked lunar impact speed and time for two different launch dates, May 30, 1966 and June 3, 1966. Figs. 4 and 5 show landing site charts for the opening of the launch windows for the same two launch dates, Figs. 4 and 5 show the following items:

- (1) Normal impact location (incidence angle at impact, $\tau_{\text{imp}} = 0$).
- (2) Loci of off-normal impact locations as functions of constant τ_{imp} for $\tau_{\text{imp}} = 5, 10, 15, 20, 25, 35$, and 45 deg.
- (3) Sub-Earth point at impact, locus of 15-deg limb restriction (Ref. 4), and terminator at impact.
- (4) Contour lines, at 5-deg intervals, showing required high-gain antenna elevation angle at impact when antenna is used for descent TV and wide-band engineering data.
- (5) Specified landing location in selenographic latitude and longitude.

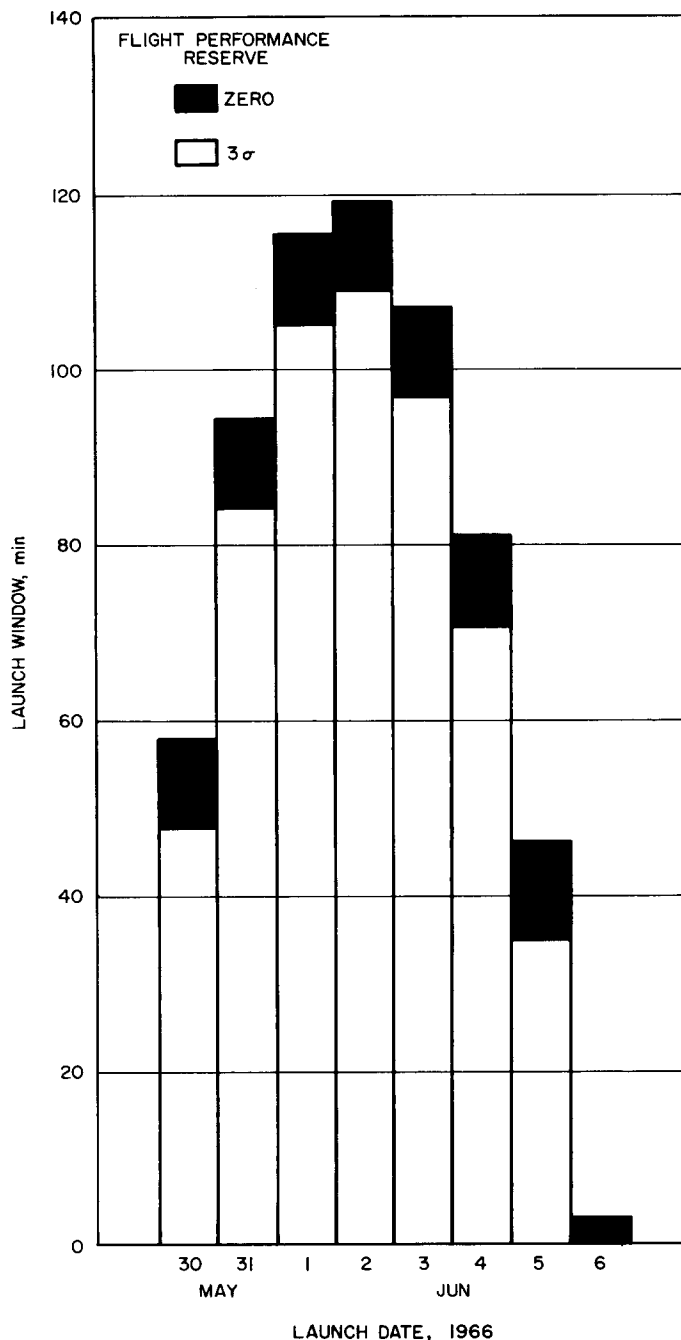


Fig. 1. Surveyor Mission A launch windows

2. Midcourse and Terminal Guidance Operational Computer Program

a. *Compensation of spacecraft + z-axis misalignment due to g-loading on sensor group during main retro burn.* This discussion describes a change in the terminal command generation program that compensates for a known error in thrust direction during main retro-rocket engine

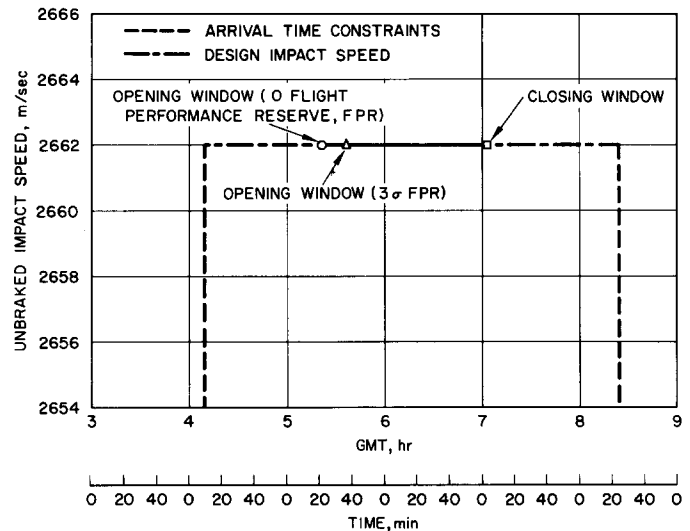


Fig. 2. Unbraked lunar impact speed and time (May 30, 1966 launch; June 2, 1966 impact; 59-min, 48-min launch window)

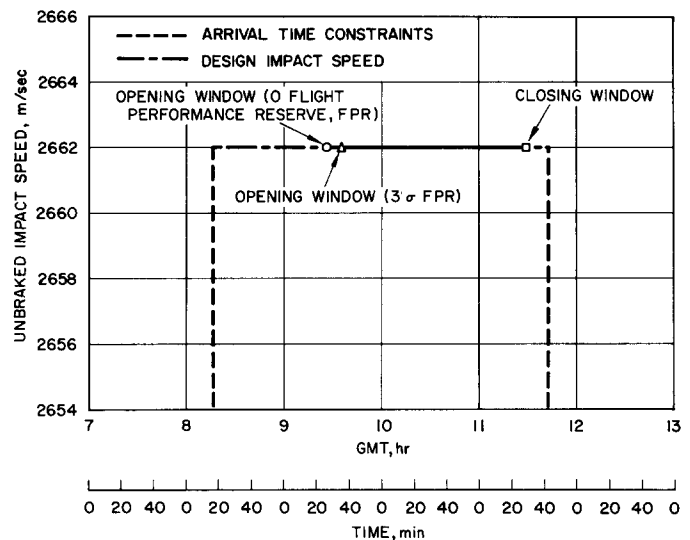


Fig. 3. Unbraked lunar impact speed and time (June 3, 1966 launch; June 6, 1966 impact; 107-min, 97-min launch window)

burn. During main retro-rocket engine burn the sensor group will experience a small angular deflection because of the g-force loading induced by the thrust of the main retro engine. As a result, an apparent rotation of the spacecraft + z-axis is sensed by the gyros and a corrective torque is applied to correct this apparent misalignment. This induces a real misalignment of the spacecraft + z-axis, which changes as a function of spacecraft acceleration.

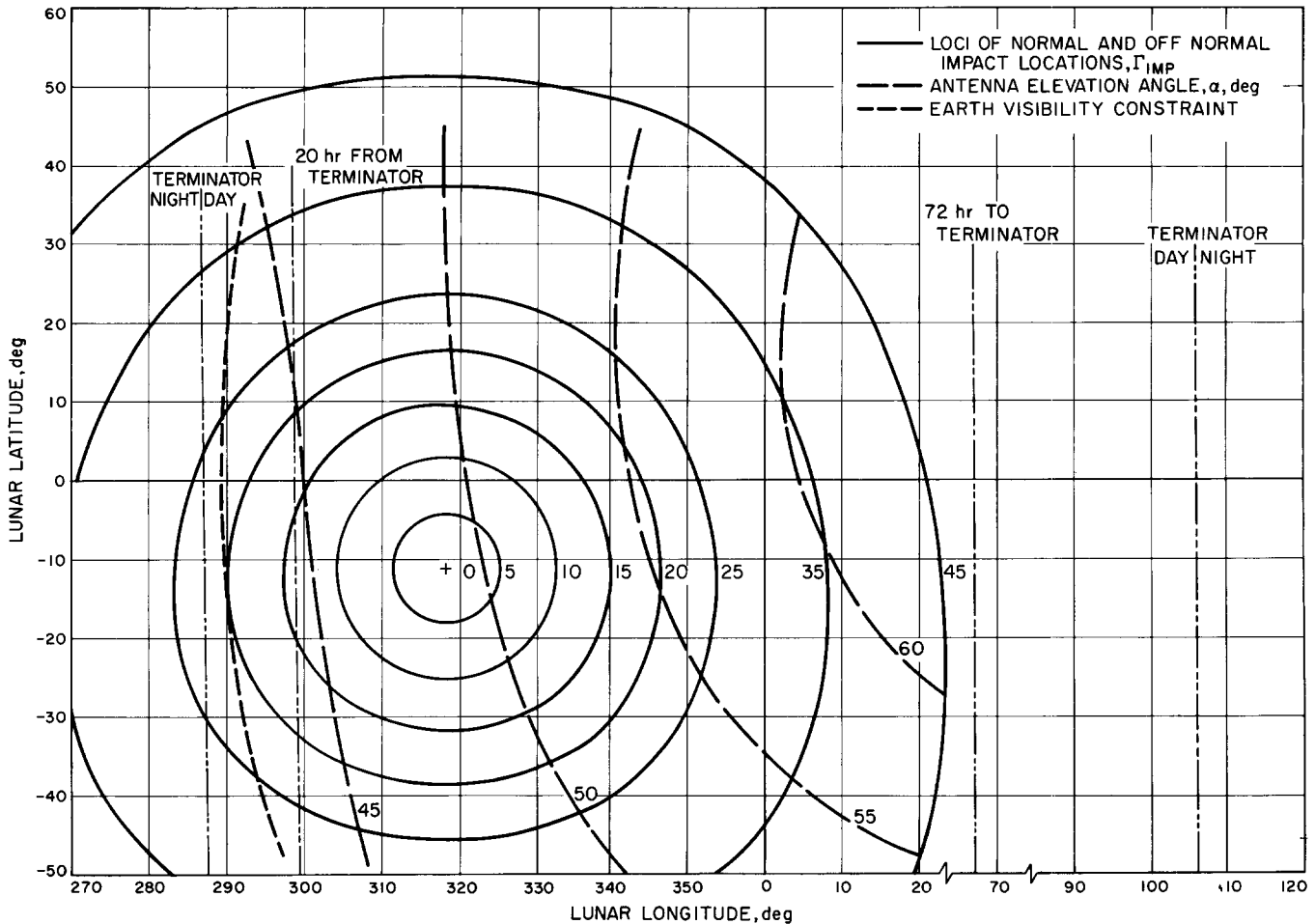


Fig. 4. Landing site chart for May 30, 1966 launch

A change in the terminal command generation program has been made to compensate for the misalignment of the spacecraft + z -axis. Pre-mission terminal analysis will supply an average misalignment angle of the spacecraft + z -axis. If the nominal thrust direction is biased by that angle in a direction dependent upon the rotation vector of the angular deflection of the sensor group, then alignment of the spacecraft + z -axis to that biased thrust direction will result in a zero total thrust direction error during the main retro-rocket engine firing phase.

Analysis. The rotation vector of the angular deflection of the sensor group is in the spacecraft xy plane at any time during main retro-rocket engine burn and is oriented at a constant angle of 65 deg from the spacecraft + x -axis directed toward the + y -axis. The average misalignment angle is then the magnitude of this vector (Fig. 6). For the purposes of rotation, denote the orientation angle and magnitude of the rotation vector by Φ and Δ , respectively. With reference to an inertial system, let the

spacecraft xyz coordinate directions during coast phase be indicated by:

$$x = (x_1, x_2, x_3)$$

$$y = (y_1, y_2, y_3)$$

$$z = (z_1, z_2, z_3)$$

Furthermore, let $[T]$ be a transformation matrix associated with a given terminal attitude maneuver sequence which transforms the coast phase attitude of the spacecraft to its ignition attitude. At this point the spacecraft + z -axis is aligned to a pre-computed nominal thrust direction. Using a primed notation, the spacecraft nominal coordinate directions at ignition are given by:

$$\begin{bmatrix} x'_1 & x'_2 & x'_3 \\ y'_1 & y'_2 & y'_3 \\ z'_1 & z'_2 & z'_3 \end{bmatrix} = [T] \begin{bmatrix} x_1 & x_2 & x_3 \\ y_1 & y_2 & y_3 \\ z_1 & z_2 & z_3 \end{bmatrix}$$

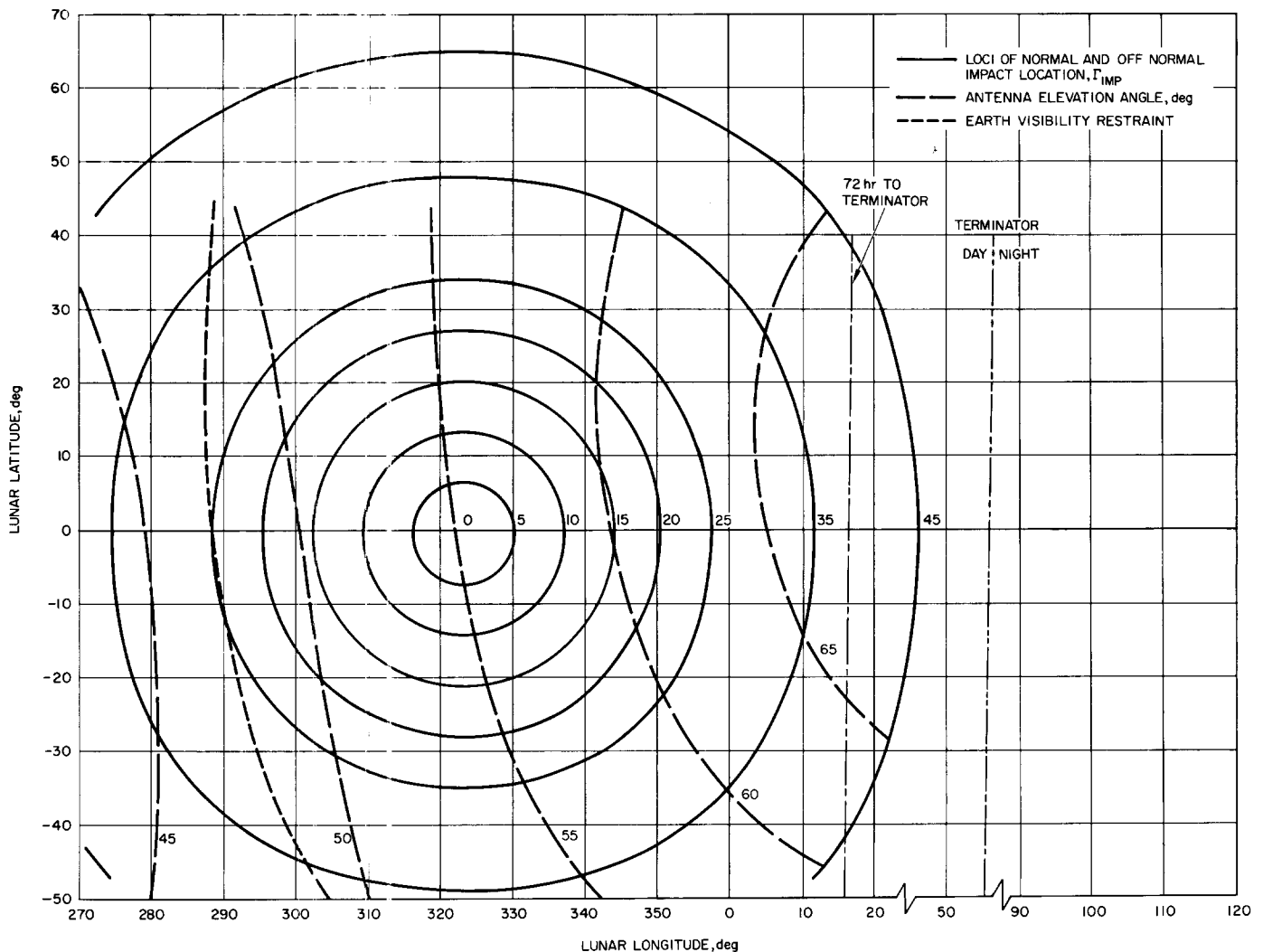


Fig. 5. Landing site chart for June 3, 1966 launch

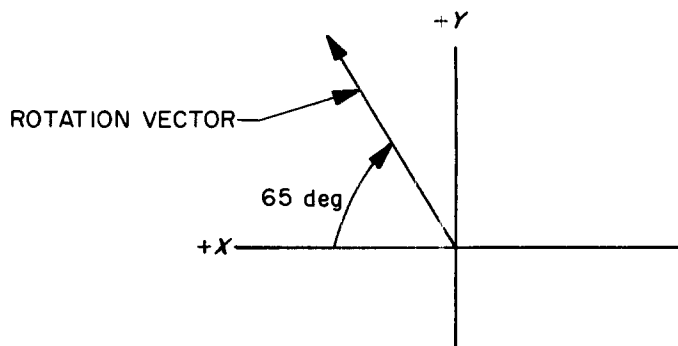


Fig. 6. Spacecraft xy plane

Correction for the misalignment error is effected by a roll transformation of PHI degrees about the z' -axis followed by a pitch transformation of DETA degrees. If the

coordinate directions after these transformations are indicated with a doubly-primed notation, the following results are obtained:

$$\begin{bmatrix} x''_1 & x''_2 & x''_3 \\ y''_1 & y''_2 & y''_3 \\ z''_1 & z''_2 & z''_3 \end{bmatrix} = [P] [R] \begin{bmatrix} x'_1 & x'_2 & x'_3 \\ y'_1 & y'_2 & y'_3 \\ z'_1 & z'_2 & z'_3 \end{bmatrix}$$

where $[P]$ is a pitch transformation of DETA degrees and $[R]$ is a roll transformation of PHI degrees.

The z'' coordinate direction is then the biased thrust direction. Alignment of the spacecraft $+z$ -axis at ignition to this z'' direction will compensate for the thrust direction error incurred during main retro-rocket engine burn because of the g -force loading on the sensor group.

These computations have been added to the terminal command program as a subroutine to compute a new thrust direction which biases the nominal direction to compensate for the error described above. Fig. 7 is a basic flow diagram of the terminal command generation program, showing the location of subroutine GLOAD in the general flow.

b. Operational computer program for computation of Earth and Sun look angles and final roll orientation of spacecraft at touchdown. For pre-mission and real-time analysis, additional information related to the terminal descent phase of the *Surveyor* mission has been requested. Three computations are involved:

- (1) Earth and Sun look angles in spacecraft coordinates at burnout and at touchdown.
- (2) Earth and Sun direction cosines in spacecraft coordinates of touchdown.
- (3) Nominal roll orientation of spacecraft at touchdown.

A Fortran II computer program, subroutine LANGLE, has been written which makes the required computations. The inputs are obtained from the terminal guidance and command generation programs. The subroutine has been incorporated into the terminal command generation program, so that the computations will be automatically made in the operational program (*Surveyor* midcourse and terminal maneuvers) used at the JPL Space Flight Operations Facility. Fig. 7 references the use of subroutine LANGLE to the overall program.

3. Lunar Radar Reflectivity Model

The revised Muhleman reflectivity model, authorized to be used for design of the *Surveyor* altitude marking radar and the radar altimeter and doppler velocity sensor,

has been defined. Previous material is affected only by the revised amplitude coefficient, which is independent of incidence angle.

The recent revision to the Muhleman model affects only the angular insensitive amplitude coefficient; the angular dependence function remains physically sound, mathematically rigorous, and experimentally justified. The derivation of this function is described fully in Ref. 6. Salient features are: (1) the assumption of certain probability distributions for displacements of lunar surface elements; (2) derivation of the density function for orientation of normals to these elements; (3) development of the surface back-scatter function, assuming geometrical optics and the previous distributions; (4) convolution with appropriate pulse lengths to derive the total hemisphere pulsed radar response functions; and (5) direct comparison with observed radar data, all in terms of one single wavelength dependent parameter representing the mean slope of the physical surface. Surface element size need not be specified, and does not enter the derivations. Assumed distributions of element displacements are gaussian in orthogonal horizontal directions (Rayleigh in one horizontal dimension), while both gaussian and exponential models were developed for vertical displacements, the latter affording a somewhat better experimental fit. Perhaps the most notable of all its features is that it includes, in a single form, the proper behavior over the full range of incidence angle variation, including the characteristically strong return near vertical. The old Lambert treatment, however, requires the addition of a specular component of proper magnitude and decay constant.

The revised coefficient ($\eta K/\alpha^3$) is that wavelength dependent factor needed for proper normalization of the integrated return for the total hemisphere. The original basis of a total radar cross section (η) equal to approximately 0.075 of the physical cross section (projected planar

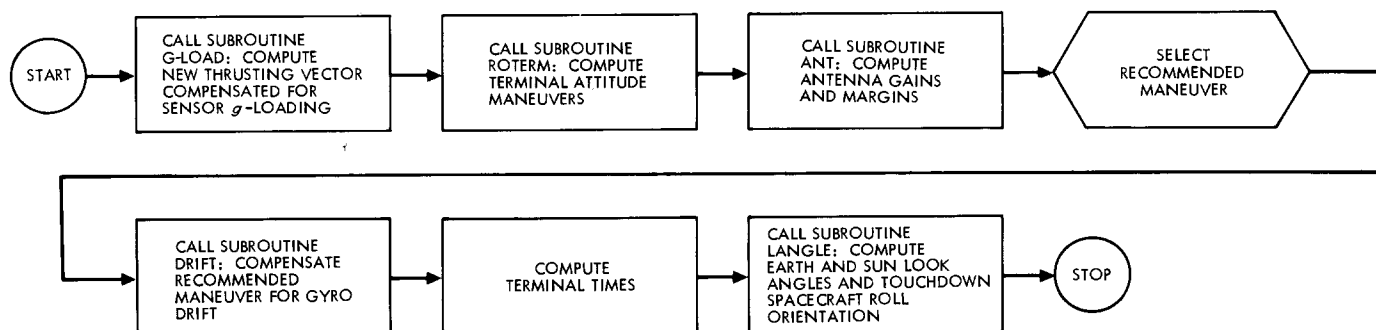


Fig. 7. Basic flow of terminal command program

disc) is unaffected, and is still utilized. This value is the most probable total hemisphere mean value. The most recent Evans and Pettingill data are less pessimistic than their former data, which had long been the major downward influence on all other data. Their recent (June 1965) data are still lower than 0.075 (by almost 3 db), but another source is higher than 0.075; hence the above conclusion is consistent with a number of other results. It is noted that Muhleman's original figure, 0.075 ± 0.05 , was intended to encompass uncertainties in the total Moon mean value, and that the reduced spread of various data (with the revised Evans and Pettingill results) has therefore also reduced that uncertainty in η as an average total value. This is quite different, however, from terrain variation with high resolution.

Scintillation, or the signal fluctuation inherent in Rayleigh scattering, can be accounted for directly as pulse-to-pulse amplitude fluctuation in the altitude marking radar. It appears adequately encompassed by proper representation of the received doppler spectrum, whose inherent fluctuation is treated as "theoretical noise" in the radar altimeter and doppler velocity sensor. Terrain variation, on the other hand, is variation of the effective radar cross section of a small fraction of the lunar surface as observed with high resolution (from relatively short distances with narrow beamwidths). While it is probably only different frequency regions of a single composite fading spectrum, distinction of the terrain variation may be related to a system time constant, with terrain variation representing the steady-state (albeit temporary) amplitude in terms of effective radar cross section, and scintillation being the "theoretical noise" in the system signal.

The amount of margin that should be allowed for terrain variation is not so readily established. Dr. Muhleman has quoted some recent radar work that achieved lunar resolutions of about 50 km². Where the five young craters are involved, increased return has been noted, that from Tycho being about 10 db above average. It is debatable if variations downward from average may be of smaller magnitude, although this possibility is supported by the absence of moisture and vegetation (normally causing weakened return in Earth data). The fundamental form of the revised lunar model calls for recognition of both upward and downward variations from average by ± 6 db (required), ± 10 db (desired).

The revised Muhleman lunar radar back-scatter function is now written as

$$\begin{aligned}\eta F(\theta) \cos \theta &= \eta K (\cos \theta) (\sin \theta + \alpha \cos \theta)^{-3} \\ &= (K/\alpha^3) (\eta \alpha^3 \cos \theta) (\sin \theta + \alpha \cos \theta)^{-3}\end{aligned}$$

Hence, the revised function is (K/α^3) times the function previously used. Proper normalization of this function is established by the integral condition

$$2K \int_0^{\pi/2} (\cos \theta \sin \theta) (\sin \theta + \alpha \cos \theta)^{-3} d\theta = 1.0$$

Numerical evaluation shows that

$$\begin{aligned}\eta (K/\alpha^3) &= -0.55 \text{ db, for } \alpha = 0.36 \text{ (appropriate for the} \\ &\quad \text{altitude marking radar)} \\ &= -1.10 \text{ db, for } \alpha = 0.39 \text{ (appropriate for the} \\ &\quad \text{radar altimeter and doppler velocity sensor)}\end{aligned}$$

with variations, in each case, of ± 6 db, required, ± 10 db, desired, as previously noted.

Apparent inconsistencies between the revised Muhleman model and the Lambert model used for the RADVS have since been explained on the basis of presence or absence of margin for terrain variation, when related to the fundamental $\sigma = 0.075 \pi a^2$ used by both. The (Lambert) $R_f = 0.005$ includes such a margin; the Muhleman $0.075 (K/\alpha^3)$ does not. To avoid confusion, the latter value is now being quoted, with separately identified terrain margin, as already discussed. Quantitatively, the margin inherent in the (Lambert) $R_f = 0.005$ appears to be 5.58 db at 0 deg (vertical) for specular 9.0 times diffuse, 5.06 db at 45 deg, and 7.06 db at 70 deg. These values (RADVS procurement specification is for the Lambert model) compare favorably with the -6-db terrain margin requirement now being used for the Muhleman model evaluation.

D. Launch Vehicle Integration¹

1. Spacecraft/Launch Vehicle Integration Schedule

a. Support of Atlas/Centaur R&D flights. Hughes Aircraft Company (HAC) completed delivery of support equipment for the Atlas/Centaur-8 (AC-8) space vehicle to General Dynamics/Convair (GD/C) December 15, 1965. The equipment, consisting of the S-band transponder and omniantenna with mounting hardware, electrical connectors, and cabling, was installed on the mass

¹Prepared by JPL Technical Section 291.

model (simulated mass of the *Surveyor* spacecraft), and a flight acceptance test (FAT) was successfully completed December 20, 1965. Completion of AC-8 systems testing at San Diego, during which time HAC personnel will conduct transponder operations, is expected during the early part of January 1966. HAC support is again required for operations involving the transponder at the AFETR, presently planned for February and March 1966.

Funding for the support of AC-9 operations has been received. All equipment to be furnished by HAC is available on call. Delivery schedules have not yet been determined.

b. Systems test and operations. Revamped test and launch schedules are being developed in accordance with *Surveyor* project redirection received from NASA headquarters December 10, 1965. Completion of AFETR operations for Mission A is now scheduled during the launch window which opens May 29, 1966. This delay has resulted in a Lewis Research Center (LeRC) decision to return the AC-7 vehicle, which has been erected at AFETR since mid-September 1965, to San Diego for updating and retest. AC-10, now undergoing acceptance testing at GD/C, has been reassigned to Mission A Spacecraft-1 (SC-1) and AC-7 assigned to Mission B (SC-2).

The new launch schedules call for a combined systems test (CST) at GD/C prior to AFETR operations for all space vehicles. The shipping date from HAC for SC-1, currently planned for February 17, 1966, results in a CST period which is presently incompatible with launch vehicle requirements; i.e., following completion of the CST and shipment to AFETR, there is insufficient time for completion of *Atlas/Centaur* operations prior to first-mate with the spacecraft. Proposed solutions are now under examination. Establishment of a work schedule, which will enable performance of a meaningful spacecraft/launch vehicle CST and provide sufficient launch vehicle AFETR time prior to first-mate, is expected during early January 1966.

Generation and revision of documentation applicable to SC-1/AC-10 CST and AFETR operations is in process.

2. Interface Requirements

LeRC comments to Revision 2 of the *Surveyor* Project Document No. 1 have been resolved to the mutual satisfaction of LeRC and JPL.

The signature page of the document was signed by the *Centaur* Project Manager and the *Surveyor* Project Manager on December 14, 1965. Cosigning was accomplished later than previously anticipated resulting in additional delay in publication. Revision 2 was released on December 29, 1965.

An agreed upon procedure for control of changes to the document has been developed. Under this procedure, changes will be issued in the form of amendments to the applicable portions of the document. Written concurrence of the *Centaur* project manager and *Surveyor* project manager is a prerequisite.

The first change leading to an amendment concerns the injection accuracy figure of merit. A draft of the proposed accuracy requirements has been approved by the *Centaur* project manager of LeRC. The proposed requirements specify launch vehicle injection accuracy in terms of the *rss* value of the spacecraft midcourse correction, assuming 1- σ injection errors.

The *rss* value is defined to be the square root of the trace of the midcourse covariance matrix computed for each nominal trajectory that may be flown and for a midcourse correction time of 20 hours past injection. Two *rss* values are used: (1) miss-only value (rss_M) which assumes that the midcourse will be used to null longitude and latitude errors on the surface of the moon disregarding errors in flight time, and (2) miss-plus-time value (rss_{M+T}) which assumes that the midcourse will correct for errors both in miss and time of flight. The requirements are stated as follows:

For engineering missions (A, B, C and D) it is required that rss_M be less than 16.7 meters/second for the trajectories considered for each mission.

For scientific missions (E, F and G) it is required that rss_{M+T} be less than 16.7 meters/second for the trajectories considered for each mission.

A midcourse figure-of-merit (FOM_M or FOM_{M+T}) is defined as the *rss* value of the spacecraft midcourse correction using the specified worst case $K^{-1}U$ matrix. The analysis leading to the selection of this worst case matrix is given in JPL TR 32-389. To meet the preceding accuracy requirements, it is sufficient to show that the FOM_M be less than 16.7 m/sec for engineering missions (A, B, C, D), and that the FOM_{M+T} be less than 16.7 m/sec for scientific missions (E, F, G) using a worst case set of injection errors.

Using the K^+U matrix contained in the proposed amendment and the latest error models, JPL has obtained an $FOM_M = 9.7$ m/sec and an $FOM_{L,R} = 12.0$ m/sec. In view of the recent change in Mission Plans calling for Missions A through G to be engineering flights and with the delay in launch schedule for Mission A, which permits the utilization of the improved guidance system, the proposed accuracy requirements will be re-evaluated prior to publication of the amendment.

Another change, which is expected to be approved soon, concerns the radio frequency interference (RFI) and electromagnetic interference (EMI) requirements. As a result of extensive testing, a set of criteria has been developed and verified. SPS 37-36, Vol. I, pp. 40-44, discusses the subject in some detail.

3. Surveyor/Centaur Separation System

Component testing of the pin-puller separation mechanism has continued at GD/C in an effort to minimize outgassing of the squib device. Through improvement of seal design, satisfactory gas containment has been accomplished.

Additional minor design changes are being accomplished in the separation spring/separation rate potentiometer assemblies to eliminate the potential problem of spring plunger hang-up, reported in SPS 37-36, Vol. I, p. 41. Precautionary procedures are being developed that will help to insure against inadvertent mishandling which could cause improper functioning of the devices in flight.

4. Surveyor Cleanliness and Contamination

Considerable effort has continued in the area of cleanliness, both from the point of view of definition of requirements and of evaluation of existing conditions.

A definitive, quantitative cleanliness specification for Surveyor has not been developed. Problems are encountered when attempting to determine the size and number of particles which can be permitted without severe degradation of various subsystems. Surveyor systems department is attending to this problem.

LeRC and GD/C have been proceeding to make improvements in cleanliness control of flight hardware, which can have an effect on the spacecraft (e.g., nose fair-

ing, adapter). Through cooperation with JPL, techniques are being considered which will enhance the cleanliness of the nose fairing and adapter at the time of final encapsulation both at CSTS (Combined Systems Test Stand) and AFETR. Attention is being given to paint problems in the nose fairing. Potential outgassing of the interior of the nose cap will be contained through a pressure bulkhead seal arrangement. This will prevent smoke from reaching the spacecraft surfaces. A problem with chalking of the fairing interior surface is presently under study.

5. Umbilical Ejection

At the request of LeRC, a study of Pad 36 umbilical requirements for Surveyor was made in support of a proposal for an early umbilical ejection. The proposal is for ejection of the upper Centaur umbilical, through which most Surveyor lines are routed, at a time of 60 sec before launch and a hold capability of 30 min at $t - 5$ sec. It was determined that, in order to accommodate the proposed timing, two Surveyor functions, main power switch control and spacecraft battery charge, would have to be rerouted to the 2-inch rise umbilical. These conclusions were submitted to LeRC.

6. Separation Sensing (SS) and Arming Device (AD)

Recent study of the Surveyor SS and AD revealed that problems exist in the mating of the device to the pedestal located on the Centaur adapter.

The SS and AD is a series of switches which actuate upon physical separation of the spacecraft from the Centaur adapter. The switches, in turn, arm pyrotechnic circuits and initiate automatic solar panel deployment. While the spacecraft is mated to the adapter, the switches are held in safe position by a pedestal located on the adapter. It was determined that the switches within the SS and AD would most likely actuate under vibration environment during boost, due to the relative motion of the spacecraft and the adapter and the small deflections necessary to cause actuation.

A two phase modification is necessary to correct the problem. An expedited fix will be developed for SC-1. A permanent modification will be developed for subsequent spacecraft. Although the details of the changes are not yet determined, some modification of the pedestal interface will undoubtedly be necessary.

7. RFI/EMI

All RFI/EMI testing scheduled to date has been successfully completed, except RFI/EMI test No. 4 with SC-1. Analysis of test data has verified that the GSE and ETR induced noise was less than the specified spacecraft susceptibility levels. It has been recommended that Test 4 be performed at CSTS with SC-1 (in view of the latest schedules) to allow time for analysis and corrective measures, should they be indicated.

8. Electrostatic Charge

Evidence obtained from ground testing on *Surveyor* and flight testing on other programs indicates that *Surveyor* could have an electrostatic charge/discharge problem with the nonconductive nose cone during the early portion of flight. A problem of this nature could result in catastrophic damage and malfunction in the spacecraft before separation.

A special committee has been formed to analyze the electrostatic charge problem on *Surveyor* and recommend corrective action. Main emphasis will be on preventive measures which can be applied to SC-1 (and subsequent), such as conductive thermolag coating on the nose cone. A testing program to be conducted at LeRC, Cleveland, is being planned.

9. Toxic Gas Detection Tests

Testing of the HAC supplied toxic gas detection system was completed at AFETR on December 15, 1965. These tests were conducted using a flight type *Surveyor* nose fairing, a dynamic model of the *Surveyor* spacecraft and the HAC mobile air conditioning system. Varying quantities of spacecraft vernier propulsion fuel and oxidizer were injected into the nose fairing cavity, and readouts were obtained on the detector apparatus. Although test data have not been completely analyzed, the correct locations of detector pick-up tubes within the nose fairings were verified, and scale readings were bracketed for determining the magnitude of fuel and oxidizer leaks. Appropriate test results will be added to the spacecraft prelaunch abort procedures.

10. Joint Encapsulation Procedures

Development of these procedures, as described in SPS 37-36, Vol. I, p. 43, is continuing. Completion of the HAC portion is expected by early January 1966. Launch vehicle personnel and representatives from HAC and JPL will then meet to generate the joint procedures for publication by GD/C.

11. Centaur Retromaneuver

The *Centaur* retromaneuver system has been modified to avoid possible impingement of the spacecraft by solid hydrogen particles which can form in the vent lines during propellant blowdown. All vehicles will carry four 50-lb axial thrust hydrogen peroxide rockets. The *Centaur* stage will begin a 180 deg turn 5 sec after spacecraft separation. At about 90 deg orientation to the velocity vector, two of the 50-lb rockets will be fired for 20 sec while the vehicle continues its turn around. The vehicle will then coast until separation plus 240 sec, when the separation distance will be approximately 2800 to 3000 ft., at which time propellant blowdown will be initiated for a duration of 250 sec to achieve the additional separation impulse required to prevent interference of the *Centaur* with the spacecraft Canopus tracker.

12. Launch Vehicle Payload Capability

A summary of current launch vehicle payload capability for the *Surveyor* missions, as of approximately November 1, 1965, is presented in Table 8. The data have been derived from Ref. 7, which is published under the new unclassified format. The vehicle flight sequence shown in Table 8 has been adjusted to reflect the recent revision of the mission plan, wherein the first seven missions will utilize A-21 "engineering model" payloads, and all but missions A, B, and D will use the parking orbit ascent mode.

The current status shown in Table 8 includes the following more important changes, which have been incorporated since the last reporting period

- (1) 56-lb gain on missions C, E, F, and G from 2-lb increase in specific impulse of *Centaur* RL 10A-3-3 engines.
- (2) 80-lb gain on direct ascent and 73-lb gain on parking orbit missions, based on revised *Atlas* base drag model.

(Only 50 and 43 lb, respectively, of the base drag model gains have been realized, because 30 lb have been placed in the flight performance reserve until additional flight evaluation can be accomplished.) An estimated 75-lb gain from the utilization of improved *Centaur* propellant utilization system (PUS) accuracy is pending LeRC approval of a final GD/C PU system accuracy analysis.

NASA headquarters has approved the *Atlas* SLV-3X for use on mission E and the following flights to assure parking orbit capability. The proposed SLV-3X has a

Table 8. Summary of current launch vehicle performance^a
(Vehicle flight sequence adjusted to reflect latest mission plan)

Mission	Flight	Payload	Ascent mode	Minimum burnout weight (BOW) ^b , lb	Centaur jettison weight, lb	Payload capability, lb	Spacecraft weight, lb	Margin, lb
A	AC-10	SC-1 engineering	Direct	6353	3955	2398	2178	+220
B	AC-7	SC-2 engineering	Direct	6343	3913	2430	2183	+247
C	AC-12	SC-3 engineering	Parking orbit	6532	4151	2381	2187	+194
D	AC-11	SC-4 engineering	Direct	6346	3942	2404	2186	+218
E	AC-13	SC-6 engineering	Parking orbit	6532	4148	2384	2250 ^c	+134
F	AC-14	SC-7 engineering	Parking orbit	6532	4145	2387	2250 ^c	+137
G	AC-15	SC-8 engineering	Parking orbit	6532	4145	2387	2250 ^c	+137

^a As reported in Ref. 7.
^b "Minimum BOW" equals "nominal BOW" less the "performance reserve" of 210 lb for direct ascent missions and 227 lb for parking orbit missions.
^c Spacecraft weight shown for missions E, F, and G is maximum allowable under current specification.

tank extension of approximately 60 in., booster engine thrust increase from 165 to 168K, sustainer engine thrust increase from 57 to 58K, and some increase in the specific impulse of the engines. At this time, only rough estimates are available for the payload gain which will result from use of the SLV-3X. These estimates range from about 200 to 350 lb, depending on the actual tank extension, skin thickness, and pressurization levels to be used.

E. Launch Operations

1. Air Force Eastern Test Range (AFETR)

a. Atlas/Centaur-7 (AC-7). The guidance and autopilot test was completed early in the report period. Cold flow encapsulation and flight acceptance test preparation continued.

The AC-7 prototype flight acceptance test and electromagnetic interference Test 3 were completed October 20, 1965. No major vehicle troubles were encountered. During the testing, the *Surveyor* retro safe and arm interface between Hughes Aircraft Company, General Dynamics/Convair (GD/C), and the Launch Pad Safety Officer was successfully exercised for the first time, as was the Launch Pad 36A *Surveyor* ready-to-launch Hughes-GD/C interface.

Other completed tasks included *Centaur* propellant utilization validation, pneumatic control unit *Centaur* checkout, insulation panel installation, and nose fairing preparation.

b. Facilities and equipment.

Safety console. The toxic gas detector was calibrated to support the shroud test. The first phase of the toxic gas shroud test was completed and approximately 100 hr were logged on the toxic gas detector.

Pneumatic control consoles. The pneumatic control consoles were checked (functional, pressure, and cleanliness) for low pressure and high pressure gasses and passed all phases of validation for which the results have been reported. However, Pan American Airways (PAA) has not returned the report on dew point and hydrocarbons content of the 6000-psi gas. The pneumatic control console units are capped and will not be uncapped until required by the project, or until meter calibration.

Microwave link. Engineering was completed on the installation of a 6-ft dish antenna on the roof of the Propellant Loading Building. The resulting 15-db improvement will remove the command link from marginal status experienced during T-21 operations. PAA installed an Andrews 6-ft parabolic antenna on the Propellant Loading Building, and testing of the RF link was continued.

Battery laboratory. Simulated operations tests were started to evaluate the procedures and the use of the battery test equipment. The main battery flight acceptance specification was exercised first. The laboratory has redundant equipment for performing many of the battery tests. In this evaluation, the relative merits of the use of the different pieces of equipment were compared.

Shroud test. Analysis of the data gathered in the past revealed that while the oxidizer portion of the test has yielded consistent results, some doubt exists as to the validity of the fuel test results. This problem probably stems from the puddling of fuel in the catch pans after it has been injected into the shroud; therefore, it was decided to extend the toxic gas tests (Fig. 8) for an indefinite period.

This new series of tests will utilize a more sophisticated injection system, which will allow a known rate to be introduced for a longer period (Fig. 9). A metal shield is being placed over the thermal bulkhead so that the propellants can be dripped from the simulated vernier engines without the use of catch pans. These tests require

support from Hughes-AFETR in maintaining and operating the mobile temperature control unit, safety console, and Teledyne portable toxic vapor detector unit (Fig. 10).



Fig. 9. Test sample injection

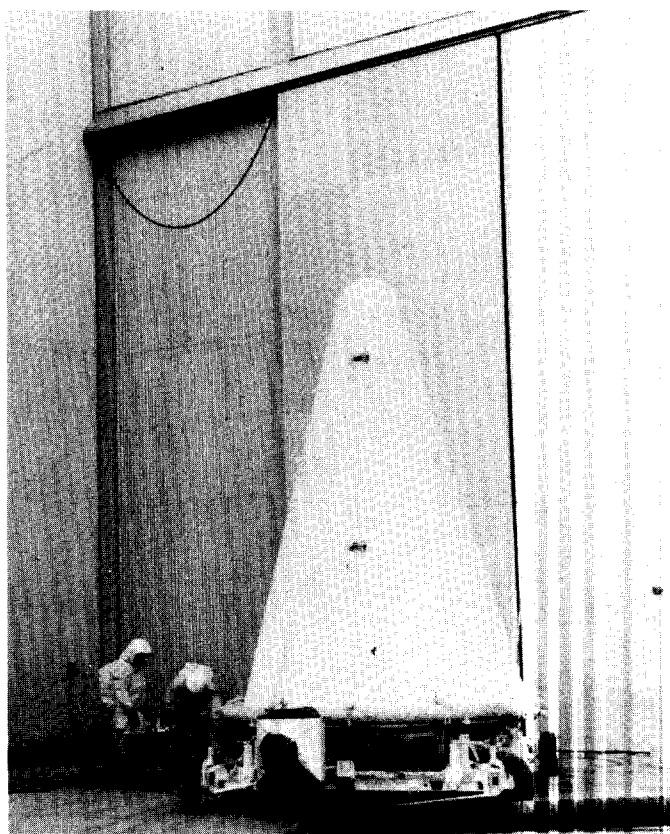


Fig. 8. Shroud test setup



Fig. 10. Portable toxic vapor detector

2. T-21 Spacecraft Operations

Weight, balance, and optical alignment procedures were performed on T-21 at El Segundo, California to verify them and to provide training for the optical dock directors (Fig. 11). All optical alignment tools were calibrated for use on T-21. Following the alignment and drop tests, T-21 will be returned to the optical dock for verification of alignment. The optical alignment, weight, and balance procedure have been revised to incorporate information gained from the test. Critiques are now under way to redirect the AFETR operations planning to reflect the changes. In these critiques, each operation will be evaluated and upgraded as necessary.

3. Mass Model Program

Mass model activities began during this report period. Hughes Aircraft Company is responsible for delivery to GD/C of one transponder each for the payloads (mass models) for AC-8 and AC-9. Hughes is responsible for test and support of the transport during prelaunch preparations at San Diego and Cape Kennedy.

The flight acceptance test of the mass model for AC-8 is to be completed in December. To support this test, Hughes has delivered transponder SD-6, an omnidirectional antenna, and various other items of hardware. Group test RF racks for transponder checkout were also

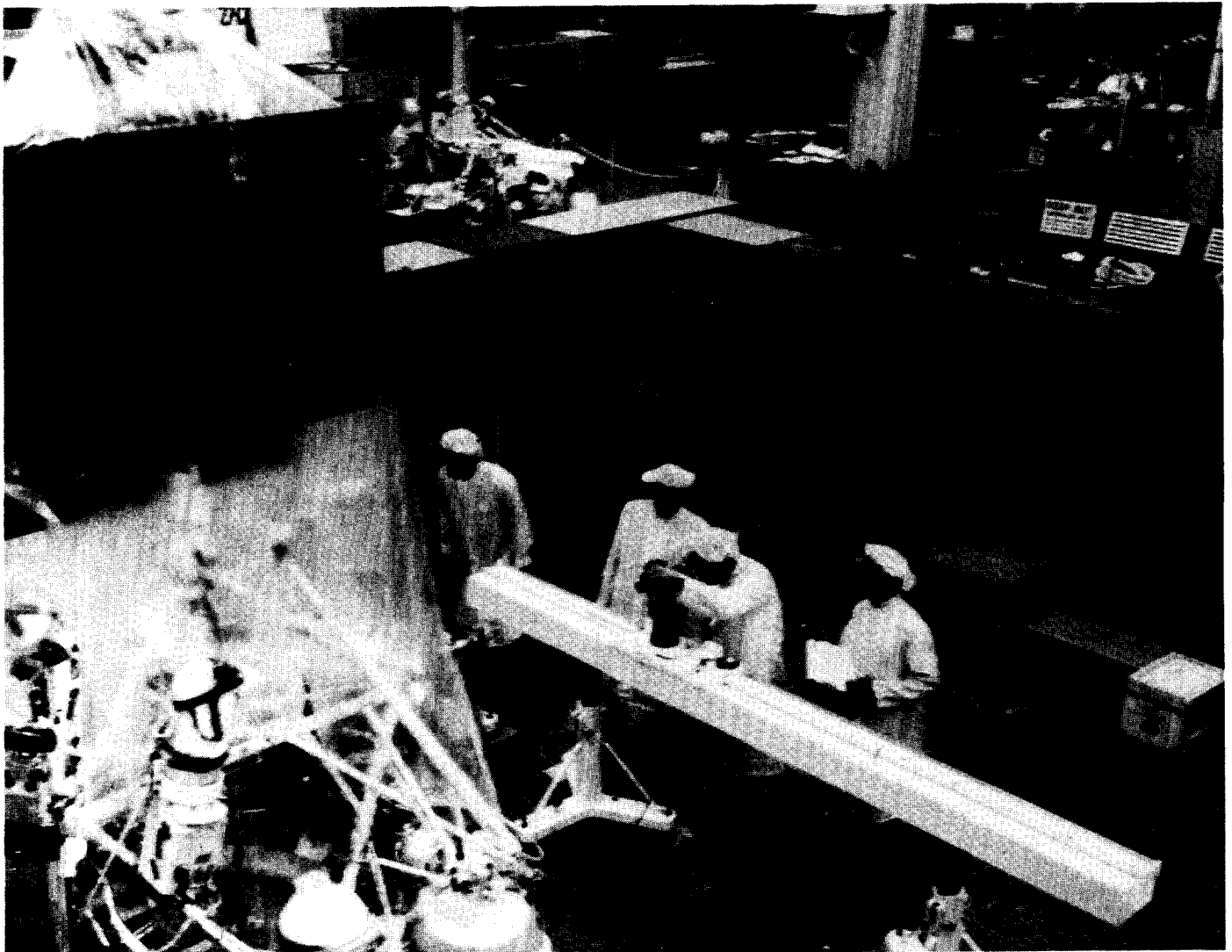


Fig. 11. T-21 weight, balance, and optical alignment operation

supplied because System Test Equipment Assembly 3 (STEA 3), which is normally located at San Diego, is now in El Segundo supporting T-21 operations.

Shortly after the start of the flight acceptance test, it was discovered that the method for mounting the transponder on the mass model (Fig. 12) represents a potential thermal problem. Preliminary thermal analysis indicates that the worst-case equilibrium temperature will exceed the upper limits of transponder design temperature.

One of the ground rules for the program calls for mounting provisions similar to SD-1 and SD-2. Provisions are similar as far as attitude and mounting pads are concerned, but the unit is positioned much closer to the spacecraft roll axis (Fig. 12). This results in the transponder bottom plate view of space being partially obscured.



Fig. 12. General Dynamics/Convair mass model

F. System Testing

1. T-21 Lunar Landing Shock Tests

The test specification, associated procedures, and test plans have been completed for the lunar landing shock test series. A change is expected in the test environmental parameters for the final drop event due to a rerun of the computer program for landing shock responses, in

order to maintain a 20.6-ft/sec velocity vector and to range the vehicle touchdown incidence angle. The landing shock absorbers were sent back to the vendor for a proof pressure test at 4200 psi to confirm an increased test safety margin. The initial drop test will be conducted with an operating flight control sensor group and radar altimeter and doppler velocity sensor (RADVS) to check turnoff capability after landing shock. Dummy units are to be available for replacement if a dummy unit is destroyed during the test. A considerable effort will be made to evaluate the telemetry phase-lock problem at lunar touchdown. This problem is thought to be more severe in the test than in an actual mission, because of the RF energy reflection problems in a steel-walled building. Preliminary tests will be conducted to evaluate performance variations due to constraints of the test location.

2. SC-1 Flight Acceptance Tests

a. Vibration tests. The specification and test plan are being reviewed for changes in requirements and procedures shown to be necessary or desirable because of hardware changes or prior test experience. Review of the test programs, from the standpoint of time conservation and overall cost reduction, is being studied.

b. Solar thermal vacuum tests. A special solar panel deployment test and a systems readiness test were run on SC-1 September 17, 1965 to verify that the failures experienced in the previously aborted mission sequence were corrected. A second systems readiness test was conducted on September 28 to investigate anomalies that were not caused by solar thermal vacuum exposure.

A systems readiness test was run October 26, 1965 and successfully completed. On October 31, a dry run consisting of primarily a midcourse velocity correction and a terminal descent were accomplished. The solar thermal vacuum exposure was started November 1 and, under a high solar condition (113%), continued for the full duration of the transit and post touchdown assessment. During this test the central command decoder did not respond to command properly, and the optimum charge regulator became erratic because of temperature difficulties. The midcourse and terminal descent were performed satisfactorily. The spacecraft is presently being upgraded to correct technical difficulties encountered in the system test, and further solar thermal vacuum tests were started in mid-December.

c. *Vernier engine vibration test.* The vibration tests, simulating vernier engine inputs, described in SPS 37-36, Vol. I, pp. 50-56 as a new test for SC-2, have been established as a requirement for SC-1.

3. SC-2 Flight Acceptance Tests

The SC-2 vehicle continued with initial systems check-out. Telecommunications, flight control, and flight control-radar tests were completed prior to upgrading the vehicle on October 29, 1965. The vehicle was returned to systems test in December. The 5-wk retest period includes vehicle calibration, design verification, and tests required to complete initial system checkout. Completion of the retest phase and MS/EMI interference test are scheduled for the next SPS report.

4. GT-1 Flight Acceptance Tests

Group testing of hardware for the SC-1 spares and the SC-3 vehicle continued during this period, as well as retesting of upgraded hardware for the SC-1 and SC-2 vehicles.

In addition, special Test 71, evaluation of external circuit modifications to the spacecraft receiver static phase error telemetry circuits was accomplished. A special test on the command and data handling console photorecorder calibration was also performed during this period.

5. T-2N Descent Test Program

a. *T-2N-1 first descent test vehicle.* On November 22, 1965, a successful descent test was performed (Fig. 13). The vehicle trajectory control during free flight followed the predicted profile. During the descent, the vehicle maintained a velocity of 5 ft/sec for 4 sec. At approximately 600 ft from the ground, after a free flight from approximately 1430 ft, recovery was initiated as programmed and all recovery functions were successful. Good test data were received on all telemetry channels and optical tracking was complete. Detailed analysis of the data is being performed, after which a complete report will be issued. At the conclusion of the test, the T-2N-1 vehicle was returned to Culver City for minor upgrade, inspection, and repeat of the system functional tests.



Fig. 13. T-2N-1 vehicle after successful descent test

b. T-2N-2 second descent test vehicle. A consent-to-test meeting was held at HAC on December 3, 1965. Permission was granted for a tether test, which was scheduled for December 14.

c. T-2N touchdown test phase. Planning has been initiated for the second phase of the T-2N program, in which the vehicle will descend under its own control for a soft landing on the ground.

G. Flight Control

1. Introduction

The flight control subsystem is comprised of spacecraft units that provide velocity and attitude control during the transit and landing phase of the *Surveyor* mission. These units, which include optical sensors, inertial instruments, attitude controls, and flight control electronics, are discussed in this section. The lunar radar and propulsion subsystems, also functionally included, are covered in other sections of this SPS.

2. Closed Loop Tests With S-6C Test Vehicle

Satisfactory performance of the *Surveyor* flight control system (FCS) operating with a live vernier propulsion system (VPS) was demonstrated during three test runs employing the S-6C flight control development test vehicle. The vehicle, located at the Placerita Canyon test facility, was tied via telephone lines into the 6-deg of freedom analog simulation of *Surveyor* flight dynamics (including flight control system and spacecraft) in Building 366 at El Segundo. The tests were controlled by tape commands generated at El Segundo and transmitted over the phone lines to Placerita Canyon; feedback from chamber pressure sensors on the three engines was transmitted back to the computer at El Segundo. Compensation for transmission line characteristics was incorporated to provide valid conditions for closed-loop system operation.

The tests were highly successful in that they validated both the performance of the integrated FCS/VPS and the method of testing. While transmission of analog data over telephone lines is a routine procedure, it is believed that closed-loop testing of a geographically distributed control system is unique. These closed-loop tests were feasible because the distances involved were short enough so that the transmission line anomalies in the range of

frequencies pertinent to the control system could be satisfactorily compensated. (It should be noted that, for a particular system, there is a limit to the distance which the system components can be separated.)

3. Flight Control Sensor Group Testing

Flight acceptance testing was completed on the SP-1 unit (235000-7 S/N 1) and delivery was made to the systems area. Upgrade operations on both the SC-1 and SC-2 spacecraft resulted in both flight control sensor groups (FCSG 235000-4 S/N2 and FCSG 235000-4 S/N 1) being returned to the subsystem test area. Functional performance tests for upgrade of various subassemblies were accomplished prior to disassembly.

4. T-2N Tests

a. Hardware modifications. T-2N flight control hardware is modified on an as-available basis to reflect the latest configuration of the SC-1 flight control hardware. Additional modifications are made in the T-2N hardware to reflect unique T-2N test conditions, such as the telemetering system.

b. Tether tests. Tether Test 3 at AFMDC demonstrated excellent performance of the inner rate loop. Transient behavior was good during engine startup and during the step from low to high thrust. Results indicated good correlation between actual and predicted system performance.

c. Descent tests. Quick look results of the first tether-descent test indicate that the flight control system controlled the T-2N-1 test vehicle within predicted limits. No vehicle or flight control system anomalies are evident. Evaluation of flight control system performance on a quick look basis is made by comparing the signals measured during the descent against analog simulations of the same signals. Prior to this descent test, extensive analog computer studies were conducted and certain critical characteristics of the descent test were tabulated for later comparisons. All possible variables of a descent test, such as release altitude, vehicle mass, measurement errors, and drag, were investigated to determine the effect of each variable on these critical characteristics.

d. Conclusions. It is concluded that the flight control performance was good. Further evaluation with optical data will be accomplished to try to resolve the cause of the small variations in the comparisons. Further descent tests planned for the T-2N-1 and T-2N-2 vehicles will incorporate upgraded flight control hardware.

5. Inertia Switch

A flight quality inertia switch has successfully withstood an extensive vibration test program. The purpose of this test program was to verify that the design is adequate to survive the predicted accumulated vibration from assembly to end of mission. The testing consisted of nine flight acceptance test (FAT) vibration exposures plus one 48-min type approval test exposure. Equating the type approval test exposure to an equivalent number of FAT exposures and adding the initial FAT shows that the unit is capable of withstanding 16 FAT exposures.

6. Inertial Sensors

a. Gyros. Serial numbers 63, 64 and 65 have been received. These are the first of the 18 gyros ordered in the latest procurement. However, S/N 63 was returned to the vendor because of out of specification non-g sensitive drift. The cause for this failure was traced to a short in shielding in the external cable.

The investigation of the decreasing transfer function in a number of gyros is continuing. As previously reported, attention is being focused on any possible changes in the fill fluid and/or changes in the damping gaps. The analysis of transfer function histories plotted against calendar time, operating time, and on-off cycles did not reveal any clear-cut correlation. However, several occurrences of the decrease in transfer function starting at the time of gyro installation into an inertial reference unit have lead to a test plan to observe the effects of the IRU environments on a control gyro.

The present FAT specification tolerance on transfer function is $\pm 10\%$, which corresponds to a value of $\pm 15\%$ assumed for the actual mission. Recent system analysis has shown that transfer function can be as much as 30% low without seriously degrading the mission.

b. Accelerometer. It was confirmed that the vibration acceleration levels applied in the previous accelerometer type approval test programs were excessive because of fixture resonances. Nonetheless, two of the four accelerometers passed the tests with no out-of-specification conditions developing; neither of the two previous out-of-specification failures was serious enough to more than slightly degrade the mission. A new vibration fixture has been fabricated, and a type approval test will be performed again on one more accelerometer after thoroughly proving out the fixture.

c. Inertial reference unit (IRU). IRU 8 was delivered and is presently allocated to the SC-3 flight control sensor group. SC-1 solar thermal vacuum testing at 115% Sun has further confirmed the adequacy of the additional radiator area added to the IRU. The lowest duty values recorded in a complete mission sequence with the 115% Sun were 2.6 and 11%.

Additional tests of low voltage on the 22-v bus have been conducted at the spacecraft (SC-1 and SC-2), IRU, and gyro temperature control module levels in an effort to duplicate a continuous temperature controller oscillation, previously observed during T-21 and SC-1 spacecraft testing. Duplication of the continuous oscillation was not obtained, although it is possible to repeatedly obtain a short burst type of oscillation. Results of measurements of the effects on gyro temperature and performance of long term exposures to the burst type of oscillation indicate that no measurable changes occurred. Circuit changes introduced in a breadboard of the temperature controller eliminated the tendency to oscillate at the low bus voltages expected. It is planned that these changes will be incorporated in the IRU designated for SC-5.

7. Flight Control Electronics Unit

a. Circuit improvements. During solar thermal vacuum testing of the SC-1 spacecraft, the retro separation squib incorrectly fired when another squib was commanded to fire. It was determined that the squib current (initiated by the command) flowing through a ground loop generated a sufficient signal to fire the retro-separation squib. The problem has been corrected by

- (1) Essentially eliminating the ground loop by making the ± 29 -v return and squib return common at the squib firing circuit.
- (2) Desensitizing the command pulse generating circuitry by adding capacitors.

The improvement realized is that a >7.5 -v step is now required to fire the squibs compared to >4 -v step before. During T-2 testing it was noted that the RADVS output was noisier than anticipated. The addition of filter capacitors at the input to the V_x and V_y modulators corrected the problem.

b. Hardware status. The flight control electronics unit, comprised of the flight control programmer and flight control electronics previously on SC-1, was delivered to the subsystem test area after being modified to include

several circuit improvements. The flight control electronics unit previously on SC-2 is being modified to include the same circuit improvements as incorporated in the SC-1 unit.

8. Gas Jet System

a. Gas supply. The tank charging valve tests planned for two Class IV tanks to verify the recommended closing torques and determine the valve leakage have been delayed by higher priority tests scheduled for the remote test station hazardous test area.

b. Attitude jets. The jet valve that was previously reported within specification after 250,000 cycles leaked 80 cc/hr after 900,000 cycles and has practically recovered after 4.67 million cycles to a leak rate of 21 cc/hr. Another attitude jet that was leaking 18 cc/hr after 260,000 cycles increased to 103 cc/hr after 1.13 million cycles, then partially recovered to 51.4 cc/hr after 2.9 million cycles. Jet valve production has been delayed because of armature chrome plating problems. A new plating source and procedure are being investigated. Armatures are under test to determine wear and adhesion qualities of the plating obtained with the new procedure.

c. System installation and testing. Manifold B nut leak tests are planned to determine the number of applications possible before leaking occurs. Also, flexible line flexure tests are planned to determine flexure life up to 500 cycles. These tests were scheduled to start early in December 1965. The SC-1 system leakage determined during solar thermal vacuum test by charging the system with argon indicates the leakage is well within allowable limits. The system, including the chamber plumbing and nonflight hardware, was charged to only 1300 psi for safety reasons. At this low pressure, the manifold that has the majority of the leak sources was close to the normal operating pressure. The jet valve leakage tests during this solar thermal vacuum test indicated little, if any, leakage. A method of determining total system gas leakage by placing the spacecraft, charged with a gas, in a plastic enclosure is being tested at Placerita on the S-6 and the S-7 test vehicles using the vernier propulsion system helium tank as a leak source.

A request has been made to modify the STEA to permit independent operation of each jet to obtain response information and also to meet other system functional

requirements. A general test procedure is being prepared and will be applicable wherever the gas jet attitude controls are tested.

9. Roll Actuator

The engineering changes for providing insulation between the pin puller on the roll actuator and the spacecraft ground have been completed. All the insulating materials have been changed to a fiberglass epoxy laminate. Vibration tests at type approval test levels have been performed on two typical installations to evaluate the design. Following the vibration exposure, the pin was retracted by firing the squib. Insulation resistance checks after pin retraction indicated satisfactory results.

The unit utilized for the pin puller insulation evaluation was the former type approval test unit. This unit has now been exposed to approximately 108 min at FAT vibration levels and 168 min at type approval test vibration levels. Except for the damage to the alignment pin on the output shaft reported previously, the unit has survived and is still functioning normally.

H. Electronics

1. Central Signal Processor

Redesign of the central signal processor is in progress. A preliminary design review was held October 10, 1965, and the final circuit release has been issued.

In the old configuration, a delay problem existed which, under worst case conditions, could cause large conversion errors at the 4400 bps bit rate. This was caused by tolerance built up in the clock pulse conditioner and binary-voltage-weighted switch turnoff time. The problem was further aggravated because the steering capacitors in the digital delay flip-flop (307) had a different time constant for charge than for discharge, thus biasing the capacitors in one direction. Also noise pulses at the input to Flip-flop 307 could cause large conversion errors. In the present configuration, steps have been taken to eliminate this problem.

2. Engineering Signal Processor and Auxiliary Signal Processor

The engineering signal processor commutator flip-flops are overloaded for worst case conditions. Low temperature tests were performed to verify the operating integrity of SC-1 engineering and auxiliary signal processor units, and they successfully passed tests at -50°C (worst case). Low temperature tests will also be performed on SC-2 through SC-4 engineering and auxiliary signal processors units. A method of reducing the gyro speed indication inaccuracies experienced with the engineering signal processor gyro countdown circuitry is currently being studied.

3. Engineering Mechanisms Auxiliary

Noise susceptibility tests were conducted on the engineering mechanisms auxiliary command and silicon-controlled rectifier power lines. The RADVS ON signal circuit was susceptible to $0.5\text{ }\mu\text{sec}$, 15-v pulses; a capacitor value change was made to extend this to $40\text{ }\mu\text{secs}$. The silicon-controlled rectifiers in the engineering mechanisms auxiliary were tested to determine their susceptibility to anode transients, and the silicon-controlled rectifiers have a rating of $50\text{ v}/\mu\text{sec}$; the *Surveyor* system, however, has an unknown value for transients induced on various power and squib lines. Test results show that a value of $2000\text{ v}/\mu\text{sec}$ was required to trigger the silicon-controlled rectifiers. A gate-to-cathode capacitor was added, and the limit was raised above $2500\text{ v}/\mu\text{sec}$, the upper limit of the test equipment.

4. Touchdown Dynamics Instrumentation Subsystem

The touchdown dynamics instrumentation subsystem provides data on spacecraft motion and loading during touchdown. To meet these requirements, the following equipment is included in the subsystem:

- (1) One inertial measurement unit (control item).
- (2) One tape recorder (control item).
- (3) One touchdown dynamics auxiliary (control item).
- (4) Three strain gage electronic units (control items).
- (5) Three leg position potentiometers.
- (6) Six contact devices.
- (7) Nine strain gage bridges.

All control items have been designed and are undergoing breadboard evaluation at the present time. Engineering problems presently under investigation include:

- (1) A more sound and definitive RFI specification.
- (2) A positive weight reduction requirement on all hardware.
- (3) A realistic development, manufacturing, and integration schedule that is compatible with spacecraft need dates.

5. Radar Altimeter and Doppler Velocity Sensor Signal Data Converter (SDC) Modification

Because the SDC was vulnerable to inadvertent noise it was decided to reduce the noise susceptibility of the SDC by modifying the tracker circuitry. A simple change was made, which consisted of adding a series resistor to the input circuit. This change has been made to X-3, X-4, and all delivered AM series, and will be included in all subsequent signal data converters.

6. Scientific Auxiliaries

Circuit design and modification tasks were accomplished on the following four scientific auxiliaries:

- (1) Micrometeorite ejecta detector auxiliary.
- (2) Single axis seismometer auxiliary.
- (3) Alpha scattering auxiliary.
- (4) Soil mechanics/surface sampler auxiliary.

The work on the four scientific auxiliaries is being handled collectively, that is, one design review and one circuit release covers all four units. The preliminary circuit release and final circuit release were issued on July 7 and November 27, 1965, respectively. The preliminary design review meeting was held on August 2 and the date of the final design review meeting was December 15, 1965.

The tasks accomplished fall into two major categories:

- (1) Protection against short circuiting the power buses.
- (2) Provision for new data handling frequencies and scale factors.

The soil mechanics/surface sampler auxiliary breadboard unit was updated and retested over a temperature range

of -40° to $+100^{\circ}\text{C}$. All HAC designed circuits operated within the specified limits.

I. Electrical Power Supply

1. Description

a. Main components. Electrical power for the *Surveyor* spacecraft is supplied by a Sun-tracking planar solar panel and a sealed, secondary silver-zinc main battery. On the first four spacecraft, an auxiliary battery will be utilized to provide additional peak load capability and increased energy storage capacity during transit. The main battery provides the electrical energy for operation after lunar landing.

b. Power management system. The power management system is composed of a boost regulator, a battery charge regulator, and a variety of voltage converters, regulators, and power switches located in other units that provide power management for various instruments and subsystems. The battery charge regulator unit provides power conversion from the solar panel, main battery charge control, and several switching functions. The optimum-charge regulator circuit provides efficient conversion of solar panel power for use in battery charging. The battery output voltage from the unregulated bus is converted to the main regulated bus voltage by a boost regulator. A main overload trip circuit removes all non-essential spacecraft loads when an overload condition is detected. The main battery charge logic system controls the charging current to the spacecraft main battery. Normally, the system operates in the automatic mode, and the main battery is charged until its voltage reaches a predetermined level. A command mode is available whereby the battery can be charged.

c. Auxiliary power. The auxiliary battery control unit provides battery voltage sensing circuitry, automatic control and selection of either the auxiliary or both batteries, and provision for command override of the automatic control mode. A main power switch provides capability to disconnect all spacecraft power.

2. Fabrication and Test Status

The first five solar panels have been fabricated. Repeat flight acceptance tests (FAT) of three of the panels is also

complete. Optimum charge regulator—solar panel integration tests were successfully conducted at Goldstone. Cracked solar cells have been found on the first three panels but small performance degradation has resulted. A change in main battery test methods has resulted in a significant improvement in battery pressure levels. The reliability program at the battery manufacturer has been completed. Main battery failure detected after SC-1 solar thermal vacuum testing is under investigation. The auxiliary battery employed in the SC-1 solar thermal vacuum test met performance requirements. Main batteries, with new separator systems for the T-2N, are undergoing qualification testing.

3. Solar Panel

The status of flight panels is as follows:

- (1) Solar panel 1—temperature sensor relocation retrofit completed. Sunlight testing has begun.
- (2) Solar panel 2—remains in storage.
- (3) Solar panel 3—repeat flight acceptance testing completed. The panel is presently being prepared for delivery.
- (4) Solar panel 4—temperature sensor retrofit completed. Initial sunlight electrical performance test and flight acceptance vibration testing completed.
- (5) Solar panel 5—final solar cell module wireup being completed.

4. Batteries

a. Main battery. As reported in SPS 37-36, Vol. I, p. 78, recent lots of batteries have exhibited a higher-than-normal pressure increase during the float portion of the HAC charging cycle. The cause of this problem has not been found, but definite pressure decrease has resulted from a change in test methods. Cell balance has been improved by incorporating an 85-hr charge and float period. The new method has been employed on the last seven batteries received and Table 9 summarizes the recorded battery pressure. A significant improvement in battery pressure has resulted. The improvement will minimize the possibility of premature charge cutoff, which could be caused by abnormal pressure increases.

The reliability program conducted at the Electric Storage Battery Company (ESB) has been completed. Examination of data indicates that the batteries performed in a satisfactory manner during simulated transit and lunar

Table 9. Battery test pressure summary

Battery serial number	Maximum pressure during cell balancing, charge cycle, psia	Maximum pressure during FAT discharge-charge cycle, psia
29	39.2	38.3
30	43.8	50.8
31	40.6	44.4
32	38.9	46.3
33	30.2	40.1
34	27.5	36.4
35	31.1	39.2

operations. A final reliability report is expected from ESB before the next SPS reporting period.

Main battery 29 was used on SC-1 for the fourth solar thermal vacuum test sequence in the first week of November. During the mission sequence, data indicated approximately 94 amp-hr were removed from the main battery, with battery performance satisfactory. At the test conclusion the battery was returned to the power systems laboratory to determine the remaining capacity and thereby verify the test data.

Initial battery inspection revealed seven possible shorted cells. The battery temperature was approximately 95°F, the pressure 51 psia, and the battery would not accept charge. The battery was disassembled and each cell pack examined. Six of the fourteen cells were partially or completely discharged. One cell showed a positive shorting path by means of silver penetration through the separator to the negative plate. No shorting evidence was found in the remaining five cells. Parallel studies are being made in an effort to isolate the failure mode.

In addition, modifications are under way in the system test equipment assembly (STEA) equipment to allow battery parameters to be continuously monitored during spacecraft use. These parameters include battery voltage, cell voltages, temperature, and pressure. Main battery 30 is being prepared for the next solar thermal vacuum test sequence for SC-1.

b. Batteries for T-2N vehicles. T-2N main batteries 122, 123, and 124 have successfully completed flight acceptance testing. The first lot of ten batteries incorporating the new separator system has been received from ESB. Qualification of the new configuration is now under way.

A heater blanket has been selected for use on the T-2N recovery battery. Qualification of the recovery battery with the blanket incorporated is in progress.

5. Central Power Control

a. Boost regulator. Three engineering changes were incorporated into the boost regulator units for SC-1 and SC-2. These changes remove the low voltage trip function of the overload trip circuit, desensitize the flight control on command to negative-going pulses, and add a filter in the *Centaur/Surveyor* separation signal input to the automatic solar panel deployment logic.

The low voltage trip function was removed to prevent turning off the nonessential bus during heavy transient load surges on the unregulated bus. Protection against negative-going noise pulses was added to the flight control on command line as a precaution against the flight control being turned on by nonstandard signals.

b. Battery charge regulator. A circuit change was incorporated into the battery charge regulators to ensure reliable operation of the optimum charge regulator at high temperatures. The units were tested satisfactorily up to 210°F in vacuum after the circuit change was made. Battery charge regulator units for SC-1 and SC-2 have been retrofitted and the upgrading of all remaining units is in process.

J. Thermal Control

The *Surveyor* thermal control system is comprised of the individual subsystems or unit thermal control systems integrated to provide acceptable thermal environments for all components during all phases of spacecraft operation. The passive and active methods employed include:

- (1) Special preparation of spacecraft external surfaces to achieve optimum thermal absorption and emission characteristics.
- (2) Superinsulated compartments to house critical equipment.
- (3) Electrical heaters.
- (4) Thermal conduction paths controlled by bimetallically operated thermal switches to maintain compartment temperatures within acceptable limits.

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Four phases of SC-1 solar thermal vacuum testing have been completed. During these tests, the vehicle was subjected to approximately 200 hr of solar irradiation at intensity levels ranging from 77 to 112% of a nominal solar constant. Eclipses ranging from $\frac{1}{4}$ to $1\frac{1}{4}$ hr were performed during each of the phases. During one of the eclipses, the vernier propellant line heaters were commanded off 5 min before simulated vernier engine firing. The test data show that this sequence results in marginal thermal performance of the vernier lines relative to the 0°F lower limit before terminal thrusting. An engineering change request has been written to alter the mission sequence to allow the vernier line thermal control heaters to remain on until touchdown.

As a result of the completed solar thermal vacuum test phases, it was determined that the vernier lines No. 3 thermostat was improperly located relative to the minimum temperature location. A new set of lines was fabricated with the thermostat relocated to the optimum position and will be evaluated in subsequent test phases. The Phase 1 temperature results indicated that the vernier Engine 2 temperatures were cooler than anticipated. As a result, the solar reflector size was increased and satisfactory performance was demonstrated in subsequent phases.

During Phase 1 high power transmitter operation, it was observed that the traveling-wave-tube temperatures were higher than desired. As a result, 0.005-in. sheets of indium foil were inserted between the thermal trays and traveling-wave-tube mounting surfaces. Subsequent thermal performance was significantly improved.

During Phases 1 and 2, Thermal Switch 6 on Compartment B exhibited anomalous thermal performance and was replaced before Phase 3. Investigation indicated a piece of foreign material embedded in the contact surface. In addition, two thermal switches on Compartment A were replaced because of badly fractured glass. It was noted that the defective thermal switches did not cause any large perturbations in the compartment performance.

Main retro-engine temperatures recorded in Phase 4 at the end of 66 hr indicated that, in general, propellant temperatures were 10 to 15° above pretest predictions. This will result in a bulk propellant temperature at the time of ignition slightly in excess of the present allowable specification. Studies to determine the potential effects of this problem have begun.

Test results from the RADVS system, the nitrogen tank, and the helium tank indicate that, in general, the resultant

temperatures are lower than the pretest predictions. While disturbing from an analytical standpoint, the resultant temperatures will not jeopardize mission performance as all systems are within their allowable specifications.

The survey camera warmed up as predicted before camera operation; however, during operation the mirror assembly temperatures dropped below the minimum operational limit of -50°F. An investigation has been initiated to determine a method of applying heater power to the camera hood assembly during camera operations.

Analytical investigations as well as thermal vacuum testing of both altimeter and doppler velocity sensing klystrons at Ryan reveal potential high temperature problems during klystron activation. The problem appears to be associated with the contact conductance between the klystrons and their heat sinks; therefore, a program of testing filler materials that will allow the optimum material to be selected for improvement of the interface thermal conductance has been initiated.

Model A-21A manifolded propellant lines have been rerouted so that portions of the lines now run beneath the lower rectangular spaceframe members. The rerouting has necessitated thermal control reevaluation and a subsequent increase in heater power requirements. The A-21A helium check and relief valves have been relocated to the upper spaceframe. The valve bodies are aluminum and adequate thermal control will result from polishing. The A-21A nitrogen tank will be partially shadowed by the touchdown dynamics instrumentation system electronics compartment; therefore, revised thermal finish patterns have been defined. The A-21A signal data converter has been located in a position very near the planar array antenna shadow line. Since dimensional tolerance accumulation or normal design evolution processes could cause the shadow line to be moved onto the signal data converter, unit relocation is under consideration. The A-21A velocity sensing preamplifier will be completely shadowed by Compartment B. As a result, alternate thermal control methods are being evaluated.

The S-10 vehicle has been shipped to JPL for testing in the JPL 10-ft simulator. The current schedule provides for testing at JPL during the month of December and at HAC during January and February.

A study has been initiated to compare the results of the SC-1 solar thermal vacuum testing with predictions made using the thermal management program. The study will

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provide a basis for upgrading the analytical parameters for the A-21 thermal management transit model. Following modification of the model parameters, the thermal management transit model will be considered in operational readiness for the SC-1 mission.

K. Engineering Mechanics

1. Introduction

The activities reported in this section include touchdown dynamics, static and dynamic structural analyses, and structural tests. Touchdown dynamics includes the specification, analysis, and testing of the soft-landing system. A significant portion of this work pertains to procurement in the landing gear shock absorbers. A similar activity is the specification of design and test parameters for spacecraft retrorocket engine separation. This section also covers stress and dynamic analyses on major spacecraft structures, the specification of loads and other design criteria, and the determination of structural test requirements. The results of the static and dynamic testing of structural models, including full-scale simulated spacecraft, are reported. This task includes procurement of flight engineering instrumentation, such as accelerometers, amplifiers, deflection potentiometers, and strain gage amplifiers.

2. Shock Absorber Design Changes and Tests

The design for the A-21A shock absorber deletes the lock finger and metering rod and provides for a removable thermal compensation bellows and a stroke increase to 5.0 in. National Water Lift (NWL) is also investigating the implications of increasing the stroke to 5.25 in., in the event another increase becomes necessary. Development unit drawings are complete and fabrication of parts is in progress.

The low-temperature drop tests and the "man rated" burst test programs have been completed and test reports submitted by NWL. Maximum loads of 10,000, 8400, and 7800 lb were sustained without failure. Gas cylinder assemblies from the drop test units were subsequently burst tested. The average corrected burst pressure was 5482 psi. All flight and type-approval test units without thermal finish have gone through an ultrasonic inspection to measure the wall thickness and are being proof-tested at 85% of bulk yield (i.e., proofed to 4250 psi).

3. Landing Dynamics

a. Effects of shock absorber rebound capability. Tests on stability capability versus lateral velocity of vehicles having shock absorbers with and without rebound on a 15-deg slope show that the no-rebound condition exhibits slightly superior stability capability. Stability capabilities versus slope at a lateral velocity of -5 ft/sec prove that the no-rebound condition exhibits slightly inferior stability capability.

4. Dynamic Analyses

a. A-21 Landing stability. Fig. 14 shows A-21 stability boundaries for minimum touchdown weight. The curves compare nominal shock absorber parameters to parameters having maximum manufacturing tolerances and temperature extremes which yield the poorest stability. Fig. 15 shows the effect of temperature on SC-1 stability. Boundaries correspond to temperature extremes (minimum crushable block temperatures with maximum shock absorber temperatures) and to landings at lunar noon and during solar eclipse. It can be seen that the stability of the maximum touchdown weight vehicle is superior to that of the minimum weight vehicle. Presently, tests are in progress to determine the lateral load capability of footpads and crushable blocks.

b. S-15 Drop test program. Preparation of the S-15 test vehicle for the A-21 touchdown dynamics study is nearing

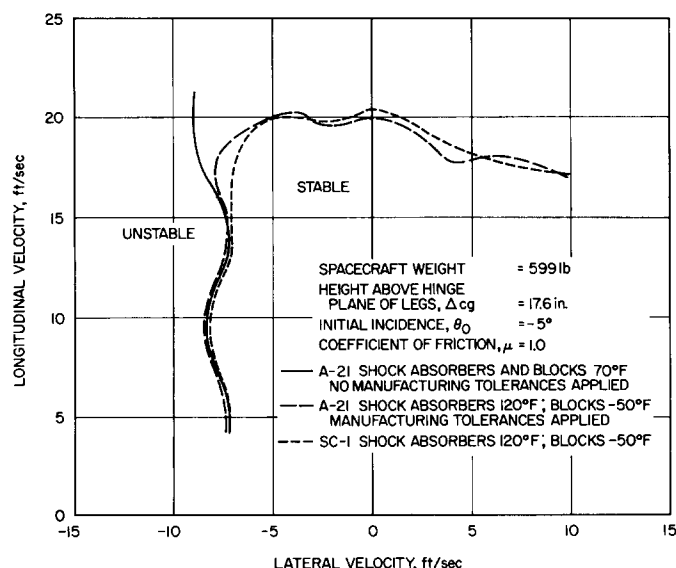


Fig. 14. A-21 Stability boundaries showing effect of tolerance and temperature extremes

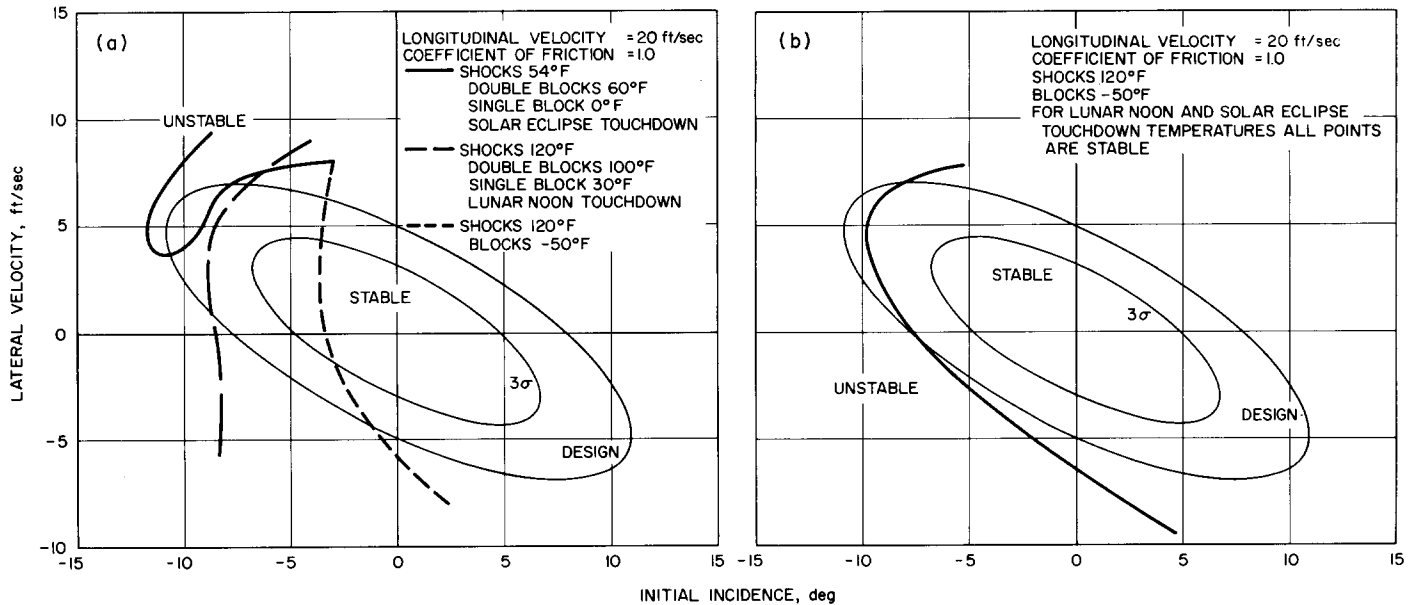


Fig. 15. Preliminary SC-1 landing stability boundaries (a) Wt., 599 lb; $\Delta cg = 17.6$ in.; radius of gyration, 32.3 in. (b) Wt., 645 lb; $\Delta cg = 16.8$ in.; radius of gyration, 31.6 in.

completion. The start of the stability drop tests may be postponed until January 1966, because the S-15 is to use the T-21 shock absorbers, and the T-21 drop tests will not be completed until that time.

Instrumentation for monitoring the vehicle angular motion just prior to and during landing has been expanded to include not only a triaxial attitude gyro system but also a triaxial rate gyro package. The additional rate information may be compared directly with rates predicted by the analytical model; whereas, the attitude data will have to undergo Euler angle transformations to allow comparison of the three-dimensional landing test results with the computer program results. Both gyro systems are ready for use in the stability drop tests.

The load-measuring support structures for the crushable body blocks have been fabricated, and strain gages are being installed. Strain gages on the shock absorbers and rigid legs of the landing gear will measure the remaining force inputs to the vehicle. To determine vehicle longitudinal and lateral velocities at touchdown, an instrumentation package is being developed which will contain two low-frequency high-sensitivity strain gage accelerometers. The acceleration data produced by these instruments prior to touchdown will be recorded, and during data reduction these signals will be electrically integrated to yield vehicle linear velocities. The successful comparison of test results and predicted results from the analytical

model is partially dependent on the accurate determination of touchdown conditions. These accelerometer signals, along with the attitude data from the gyros, should provide the necessary information.

c. A-21A Shock absorbers. A shock absorber having a constant compression damping orifice was proved to have superior landing stability to one having a varying compression damping orifice, and is easier to manufacture.

d. Impact and fall landing dynamics. Closed form equations for the gross effects of *Surveyor* lunar landings have been obtained by considering landing dynamics in the form of the separate events of impact and fall. For the impact event, an impact point is assigned to the spacecraft; for example, the foot-pad position with the shock strut at midstroke or a crushable block at midstroke. The impact equations are based on momentum relations (e.g., conservation of angular momentum about the impact point), and the limitations due to friction and lateral strength of the crushable material are included. Gross effects of the impact given by the closed form equations are as follows:

- (1) Energy dissipated during impact
- (2) Impulses of impact
- (3) Velocity and displacement conditions after impact

The fall event occurs between impacts and is the falling of the spacecraft under the action of gravity. Equations are given for each of three possible types of fall:

- (1) With contact point of vehicle stationary on lunar surface
- (2) With contact point of vehicle sliding with friction on lunar surface
- (3) Free fall with no contact with lunar surface

These equations have been used to define shock spectrums for the spacecraft. The shock spectrums are defined by the time lags between impacts and landing conditions and are categorized by groups of constant time lag, τ . These groups provide the means for coupling landing dynamics with the frequencies of the elastic spacecraft structural response. For the condition of falling with the first contact point stationary on the lunar surface, shock spectrums have been defined for landings with longitudinal velocities of 20 and 15 ft/sec.

5. Dynamic Tests

a. S-9 Structural test vehicle. Qualification level variable frequency torsion tests were performed on the S-9 vehicle at the JPL space environment test facilities on October 18. Torsion pulse tests which followed the sine sweeps were completed on October 21 and the test article returned to Hughes for a posttest inspection. The test vehicle has completed manufacturing preparations for static test. The static test setup is being prepared prior to instrumentation and calibration of the vehicle. Load application and measurements are scheduled to begin on January 17. The S-9 torsion tests are now complete. Tests are continuing on evaluating the torque/load relationship for titanium bolts into aluminum bases.

b. Crushable block and footpad tests. Preliminary tests have been conducted to determine the crushable properties of the body blocks under combined axial and lateral loading conditions. While the block was subjected to an axial crushing force, a lateral crushing force was applied through a semicircular collar at the base of the block. The test was designed to determine the ratio of lateral force to axial force as a function of the ratio of lateral crushing velocity to axial crushing velocity. Tests run to date have been for velocity ratios in the range from 0.05 to 0.22 and show that in this range the lateral force is less than 7% of the axial force. The tests will be continued to investigate higher velocity ratios.

6. Static Tests

a. Torque tests of titanium bolts for helium tank installation. The titanium helium tank has a flanged port for attachment of an aluminum valve housing assembly. Four $\frac{1}{4}$ -28 by 4-in. titanium (6Al-4V) bolts threaded into the tank flange provide the clamping preload necessary to prevent separation of the mating surfaces between the elements of the tank-housing assembly under internal pressure loading. Torque tests, to a maximum test load of 2500 lb, set the torque on the helium tank at 180 to 200 in.-lb, which corresponds to a minimum preload of 1500 lb.

L. Propulsion

Inflight propulsion is provided by a solid propellant retrorocket engine and a liquid propellant vernier system. The retrorocket engine provides primary deceleration during descent to the Moon. The vernier system is used for midcourse maneuver, attitude control during retrorocket engine firing, and terminal descent. HAC developed the feed portion of the vernier propulsion system, and the thrust chamber assemblies are provided by JPL from Reaction Motors Division of Thiokol Chemical Corporation. The Elkton Division of Thiokol developed the retrorocket engine under subcontract to HAC.

1. Status of Engines and Tests

a. Main retrorocket engine (A-21). The SC-1 flight engine is in storage at AFETR, together with an inert engine, which was supplied to facilitate spacecraft and equipment checkout. The SC-1 spare engine is in storage at Thiokol, Elkton.

b. Main retrorocket engine (A-21A). Thiokol, Elkton is proceeding with the design phase of the A-21A program. Preliminary drawings have been released for thermal and stress analysis. The composite nozzle structure was analyzed by the Thiokol, Wasatch Division, and the balance of the inert hardware by the Elkton Division. A preliminary stress report and the initial design review brochure have been submitted to HAC.

c. Vernier propulsion system (A-21). The type-approval test review committee (TATRC) granted acceptance of the S-6 and S-7 type-approval test programs. A posttype-approval test pressurized drop test was successfully conducted on S-7. Formal reports were released with regard to the type approval test and pressurized drop tests.

TATRC's acceptance of the S-6 and S-7 results and the successful drop test completed all planned activities for the S-6 and S-7 propulsion test vehicles. The S-6 propulsion system is currently being used to investigate the margin of stability by operating through a closed loop with a computer simulating the other vehicle subsystems.

d. Special support tests. Several subsystem tests were conducted on propellant tank stress corrosion, free helium formation, propellant vapor freezing, and system leakage detection methods. Test results indicated that the system as designed and tested will have no mission problems in these areas.

e. Vernier propulsion system (A-21A). All vendors have made good progress, and component-level type approval testing has begun on many components. Tests simulating lunar dumping of propellants indicate no significant degradation of thermal or optical finishes should be expected. The configuration of test vehicles S-6A and S-11 was defined. Preliminary tests of the aluminum foil laminated bladder were successful.

f. T-2N. The vernier propulsion system on T-2N-1 continues to operate completely satisfactorily in tether and descent tests.

g. AFETR operations. All propulsion aerospace ground equipment was validated in preparation for servicing SC-1. Operating procedures continue to be upgraded.

2. Main Retrorocket Engine

a. A-21. The test data for the SC-1 acceptance engine, which was tested September 16, 1965 (A21-25), are indicated in Fig. 16 which shows the vacuum thrust and chamber pressure history of the engine. The pertinent

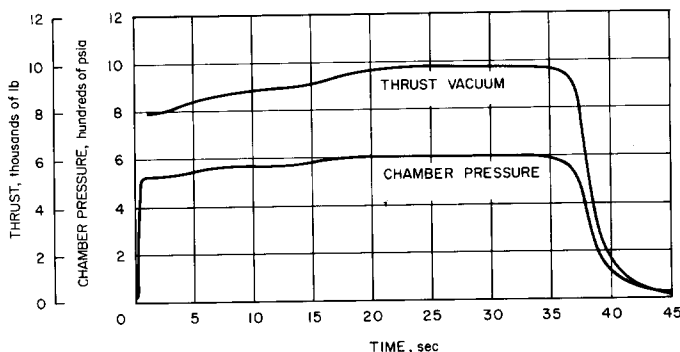


Fig. 16. Vacuum thrust and chamber pressure for engine A21-25

Table 10. Acceptance engine (A-21-25) parameters

Inert assembly weight (before test)	139.76 lb
Inert assembly weight (after test)	129.55 lb
Engine assembly weight (before test)	1377.73 lb
Nozzle expansion ratio (mean)	50.97
Test condition	Nominal gradient
Average pressure (over burn time)	582 psia
Maximum vacuum thrust	9800 lb
Total impulse (over burn time), vacuum	343,789 lb-sec
Vacuum specific impulse	289.0 lb-sec/lb
t_{3500} , vacuum	38.76 sec

physical and performance parameters for the test are noted in Table 10.

Tailoff data beyond t_{3500} were not measured directly because of test facility limitations. All specification requirements were successfully met.

The balance of the A-21 flight and flight-spare safe-arm assemblies are being processed through the HAC thermal control application procedures. The assemblies will then be flight acceptance vibration tested, reacceptance tested, and placed in bonded stores at Thiokol to await installation.

Initial deliveries of the explosive safe area retrorocket engine heater assemblies have been received and processed through inspection. Also, panels of the super-insulation blanket cover, with the conductive 304 stainless steel wire mesh installed, have been received and placed in bonded stores.

All flight retrorocket engine ignition squibs have successfully passed flight acceptance test vibration. A portion of the squibs was withdrawn and fired under over-current conditions. No anomalies were noted in firing at either 9.0 or 20.0 amp (normal firing current is 4.5 amp). Function times were of the order of 0.25 to 0.50 msec.

b. A21-A. The nozzle design and analysis effort was to be completed December 16, 1965 and presented for review. The inert weight budget for the design effort has been 125 lb. The case, nozzle closure, strut, skirt, and insulation design and analysis have been handled by the Thiokol, Elkton Division. The Thiokol, Wasatch Division conducted the analysis of the composite nozzle structure.

A schedule slip amounting to 4 weeks has been incurred because of a United Steelworkers' Union strike at the

Nevada plant of Titanium Metals Corporation, the case material vendor. Both HAC and Thiokol are attempting to establish means of reducing the slip but with little success as yet. The inert engine deliveries, scheduled for April 12 and July 15, 1966, will be the first to be delayed.

3. Vernier Propulsion System

a. A-21 system analysis. Ingestion of helium into the propellant stream can occur with the current standpipe design under certain conditions of acceleration and flow rate if a helium bubble is entrapped within the propellant tank bladder. Further study and tests as to the probable size of any entrapped gas bubble indicate that a worst case bubble would be too small to cause any problem.

b. System type-approval test program. All system type approval tests on test vehicles S-6 and S-7 have been successfully completed. The TATRC met and reviewed the test results. The performance of both vehicles was judged to be completely satisfactory. A full report documenting the test results and conclusions for the S-6 and S-7 was prepared.

c. System pressurized drop test. At the conclusion of type approval testing, the S-7 vehicle was refurbished and instrumented in preparation for a pressurized drop test. The drop test was successfully completed October 11, 1966. All systems performed satisfactorily.

d. Special support tests. Subsystem support tests were conducted in the areas of propellant vapor freezing, free helium formation, and system leakage detection methods. A section of the pressurization lines is expected to get as cold as -100°F during flight. Propellant that has permeated through the bladders could condense and freeze in this area, thereby blocking the pressurization lines. Tests on mockup lines controlled to temperatures of -100°F and colder demonstrated that the lines would not be blocked because of vapor freezing. Subsequent analysis revealed that N_2O_3 (freezing point approximately -150°F) is formed when the NO_2 and NO vapors combine. The results of these tests and analyses show that the cold pressurization lines will not adversely affect vernier system performance.

e. Further testing with S-6 vehicle. Following completion of type approval testing, some of the S-6 vehicle subassemblies were assigned for use on a flight control system verification test program.

f. Helium tank and valves assembly. Three helium tank and valve assemblies had high leak rate after flight assem-

bly test vibration. The design and procedures for assembly and test were thoroughly analyzed, and measures were taken to solve the problem.

The first assembly to go through flight assembly test after incorporation of the corrective measures completed the postvibration high pressure leak test with the lowest posttest leakage rate recorded on any assembly to date. No further difficulties are anticipated.

Helium tank and valve assembly type approval test was completed. Tests included exposure to thermal vacuum, acceleration, and shock environments, and a proof test with leak checks after each phase.

g. Propellant tank assembly. Type approval testing began November 29 utilizing two new tank assemblies of flight configuration (one oxidizer and one fuel). The vibration portion and approximately one half of the thermal-vacuum phase of testing has been successfully completed. Type approval tests will be completed next.

h. Tank shells. The test program utilized a *Surveyor* 6Al-4V titanium propellant tank previously used on the T-21 program, and consisted of removing the bladder and subjecting the tank to 7 days storage while it contained MON-10 at 850 psig and 110°F . The tank was then proof-tested at 960 psig. After proof-test, the tank was sectioned and microscopically examined for any evidence of incipient failure or stress corrosion; no evidence of either was found—confirmation that the *Surveyor* tanks are completely satisfactory for flight.

i. Thrust chamber assembly. The flight engines for SC-2 and SC-3 and the test engines for use on S-6C (the designation for the flight control system verification test vehicle) were delivered during this report period.

j. A-21A. Purchase orders for almost all components have been placed and initial delivery of several short lead time items has been completed. Activities with vendors have been closely coordinated and several trips have been made to vendor facilities to participate in preliminary design. Type-approval test specifications are complete, and type approval test is beginning on helium squib valves, regulators, tank shells, check valves, and relief valves. The majority of type approval tests at the component level are being conducted at vendor facilities with HAC quality assurance and engineering surveillance.

Design reviews on the dump squib valve and the anti-migration check valve were conducted with the vendors.

An exposure test to propellant vapors in concentrations equivalent to those expected during lunar venting was conducted on representative thermal finishes and optical samples. Property measurements before and after exposure confirm that no measurable degradation occurred on any sample.

The S-11 vehicle configuration has been defined. S-11 will be issued for dynamic environmental type approval test. No engine firing is contemplated on S-11. However, the postlanding vent system will be tested after this vehicle has been subjected to type-approval test vibration and landing shock tests.

Engineering tests of the metallized bladder were completed. The FEP-AL-TFE laminate bladder completed 10 expulsion cycles before leakage occurred. Eight cycles were at ambient temperature and the last two were at 100°F. The major advantage of inclusion of the thin layer of aluminum sandwiched in with the usual layers of TFE and FEP teflon is the greatly reduced helium and oxidizer permeability. No further effort is planned on these new bladders since the system performs completely satisfactorily with the present bladders.

k. *T-2N*. The vernier propulsion system performed as planned and in a completely satisfactory manner in the tests of T-2N-1.

l. *AFETR operations*. All propulsion aerospace ground equipment has been validated in preparation for servicing SC-1.

4. Retro Phase Moment Control Compensation Due to Vernier Propellant Usage²

a. *Summary*. During the main retro phase of the lunar descent maneuver, misalignments between the retro thrust vector and the spacecraft center of mass cause a moment to be applied to the spacecraft which is counteracted by differentially throttling the vernier engines. Since each engine is fed from a separate set of tanks near it, the differential propellant usage of the engines causes a shift of the spacecraft center of mass in a direction which partially compensates for the applied moment and reduces the corrective moment required by the vernier engines. This study indicates that vernier system compensation approximately doubles the moment control capability of the vernier system by the end of the retro maneuver.

b. *Introduction*. During the initial phase of the lunar descent maneuver, the main retro motor produces approximately 10,000 lb of thrust to decelerate the spacecraft. Small misalignments between the main retro thrust vector and the spacecraft center of mass cause a large moment to be applied to the spacecraft which is counteracted by differentially throttling the vernier engines. The moment control requirements for the vernier engine system are critical because of the limited throttling range of the engines (30 to 104 lb of thrust). Since each individual engine is fed from a separate set of tanks near it, the differential propellant usage of the engines causes a shift of the spacecraft center of mass in a direction which partially compensates for this applied moment and reduces the corrective moment required by the vernier engines.

The overall configuration of the *Surveyor* propulsion systems is shown in Fig. 17. The thrusts of the vernier engines and the retro motor are perpendicular to the plane of the figure. The object of this analysis is to calculate the magnitude of the compensation for a variety of initial conditions.

c. *Analysis*. A computer program was written to numerically evaluate the effect of vernier system compensation. It was assumed that the retro and vernier specific impulses are constant, the sum of the three vernier engine thrusts is a constant and that all engine thrust vectors are

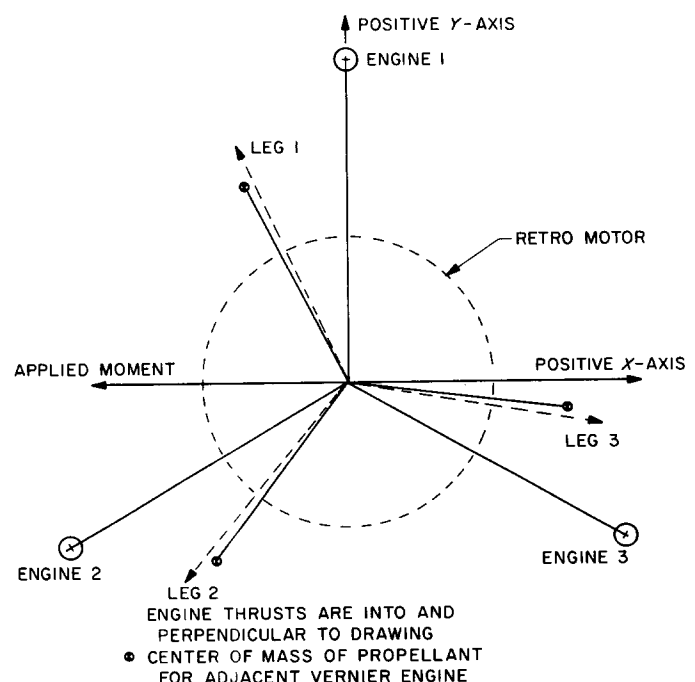


Fig. 17. Propulsion system configuration

²Prepared by JPL Technical Section 384.

perpendicular to the plane of Fig. 17. The effect of the retro motor on the spacecraft was treated as a vector perpendicular to the plane of the three vernier engines passing through the spacecraft center of mass and a moment vector lying in the plane of the vernier engines.

Using the system parameters listed in Table 11, the program calculates an initial combined center of mass location and with the input moment calculates the initial thrusts of the three vernier engines. These initial thrusts are assumed to remain constant for a time increment of 1 sec, and the amounts of propellant used by each engine during this time interval are calculated. Then the new position

of the combined center of mass is found, and this process is repeated until burnout.

d. Results. This program was used to determine the vernier system compensation for both the A-21 and A-21A Surveyor spacecrafts. Calculations were made with the spacecraft center of mass initially on the axis of symmetry and with the center of mass displaced 1 in. from the axis of symmetry. In all cases the applied moment is in the $-x$ direction, as shown in Fig. 17. For this particular study the applied moment is defined as the product of the retro thrust and the misalignment between the retro thrust

Table 11. Propulsion system configuration and performance parameters

Parameter	A-21	A-21A
Radial distance to vernier engines from origin, in.	37.8	37.8
Radial distance to vernier propellant tank centers of mass from origin, in.	25.9	25.9
Angle between vernier engines and the center of mass of their propellant tanks, deg	22.3	22.3
Main retro thrust (max), lb	9500	11,800
Main retro flow rate, lb/sec	31.1	35.3
Weight of dry spacecraft, lb	760.67	949.3
Vernier engine specific impulse, sec	275	275
Total thrust of vernier engines, lb	201	201

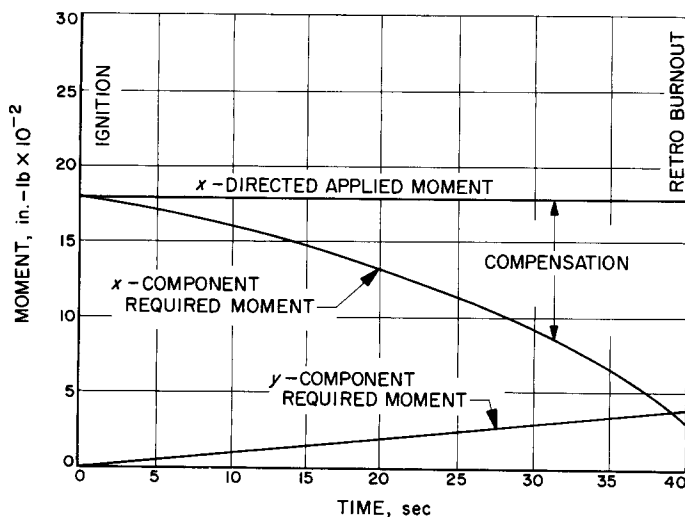


Fig. 18. A-21 spacecraft moment requirements for constant applied moment (center of mass initially at origin)

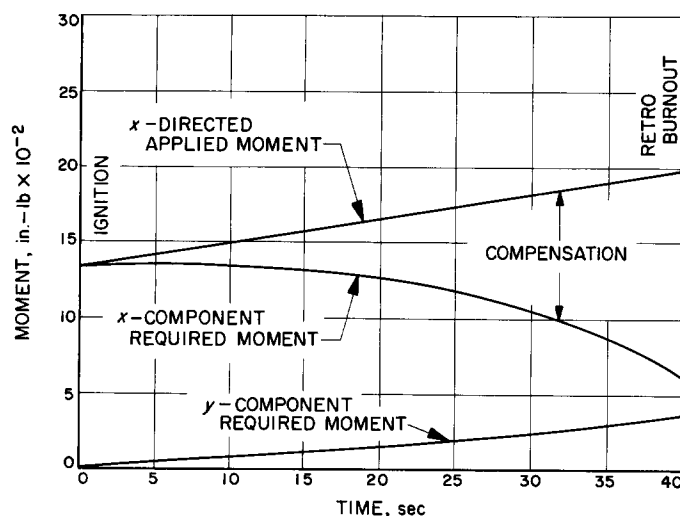


Fig. 19. A-21 spacecraft moment requirements for ramp moment (center of mass initially at origin)

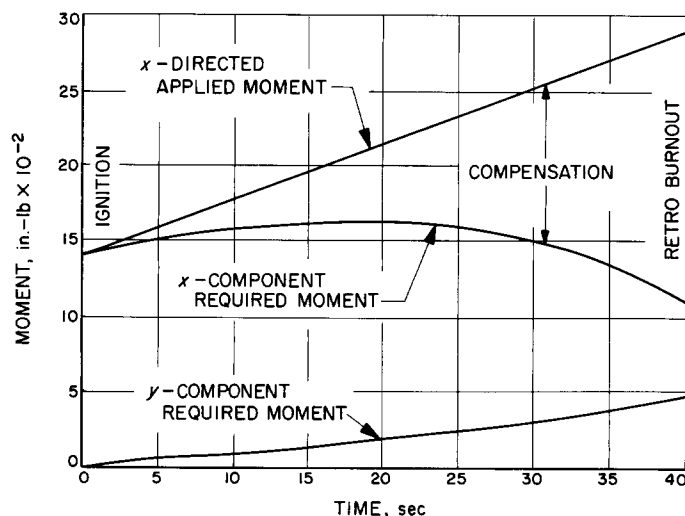


Fig. 20. A-21A spacecraft moment requirements for ramp moment (center of mass initially at origin)

vector and the initial spacecraft center of mass. Thus the applied moment is the moment which would be required if there were no vernier system compensation. Several applied moment profiles were used to evaluate system performance.

Figs. 18, 19, and 20 show the applied moment, the x and y components of the required moment, and the amount of compensation received as functions of time. Fig. 18 refers to the case of a constant applied moment, Fig. 19 to a ramp moment for the A-21 spacecraft, and Fig. 20 to a different ramp moment for the A-21A spacecraft. The initial center of mass was on the axis of symmetry.

Figs. 21, 22, and 23 show the mass of propellant remaining in the tanks of the three engines in question by comparing each propellant mass to the mass remaining in the

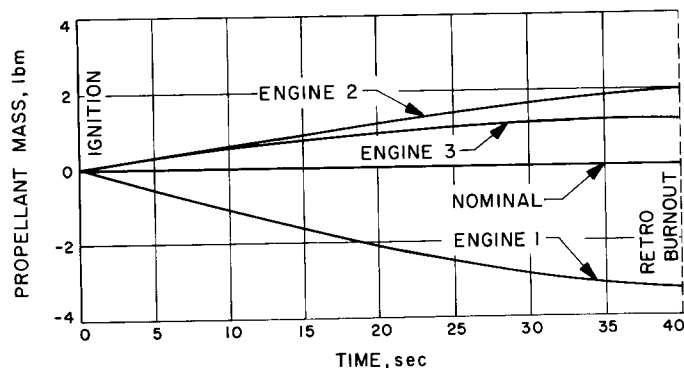


Fig. 21. Comparison of calculated propellant usage to propellant usage of A-21 engine at nominal thrust (constant applied moment; center of mass initially at origin)

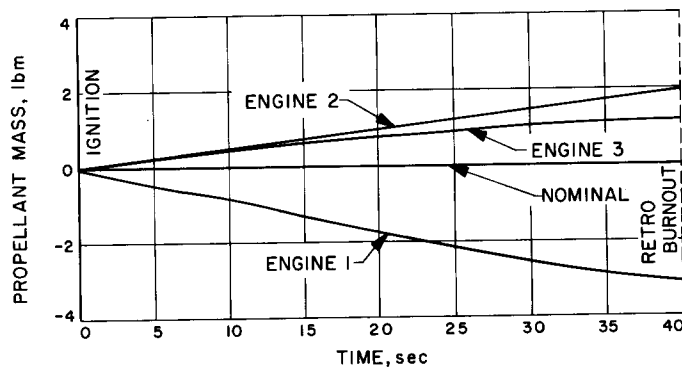


Fig. 22. Comparison of calculated propellant usage to propellant usage of A-21 engine at nominal thrust (ramp moment; center of mass initially at origin)

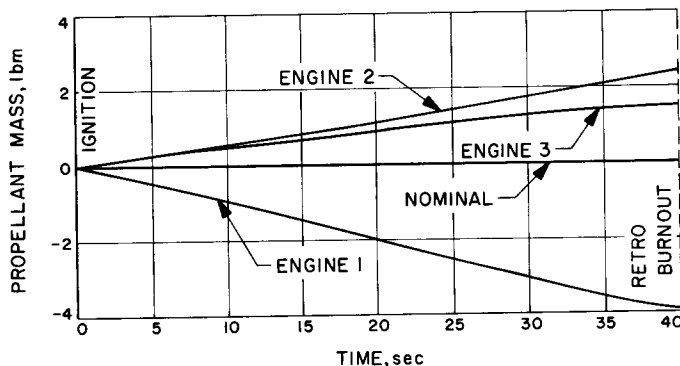


Fig. 23. Comparison of calculated propellant usage to propellant usage of A-21A engine at nominal thrust (ramp moment; center of mass initially at origin)

tank of an engine operating at nominal thrust as functions of time for the above cases. The nominal case is the zero level, a positive value indicates a usage rate less than nominal, a negative value a usage rate greater than nominal. Fig. 21 is the case of a constant applied moment, Fig. 22 is a ramp moment applied to the A-21 spacecraft, Fig. 23 is the ramp moment applied to the A-21A spacecraft.

Fig. 24 shows the applied moment, and the required moments resulting when the retro motor and the dry spacecraft centers of mass are offset 1 in. from the spacecraft axis of symmetry toward (positively) and from (negatively) Engine 1 for the A-21 spacecraft. Fig. 25 shows similar results for the A-21A spacecraft.

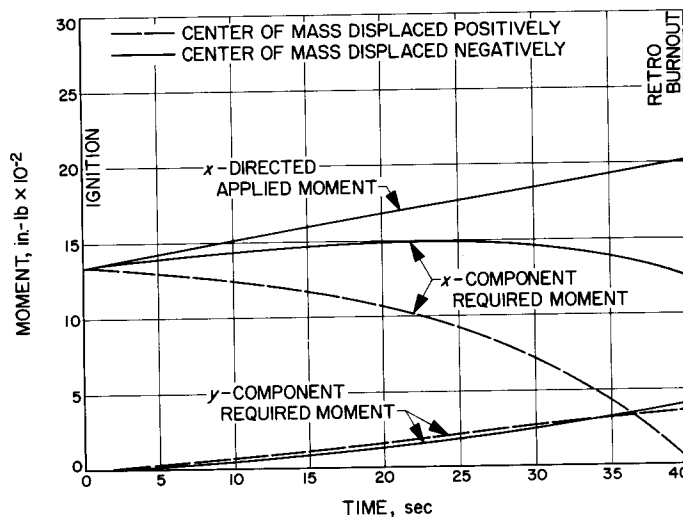


Fig. 24. A-21 spacecraft moment requirements (center of mass offset 1 in. from axis of symmetry)

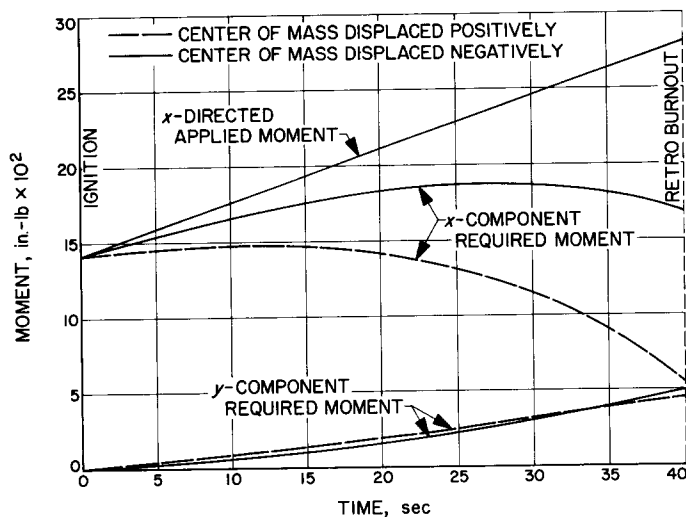


Fig. 25. A-21A spacecraft moment requirements (center of mass offset 1 in. from axis of symmetry)

e. Conclusions. The results of this analysis show that vernier system compensation greatly increases the effective moment control capability. For the cases considered, the maximum required moment was only slightly in excess of the initial moment though the applied moment doubled during the retro burning period. The moment requirements shown should be compared with a capability of at least 1880 in.-lb, the minimum design criterion. By extending this analysis, an optimum alignment can be found for the main retro thrust vector between the initial and final spacecraft centers of mass.

5. Thrust Chamber Assembly³

The evaluation of the *Surveyor* vernier engines (RMD⁴ Model No. TD-339) was continued with fifteen tests being performed at JPL-Edwards Test Station during November and December. The test summary which follows is arranged in groups according to test goals.

Group	Test No.	Test date	Purpose
12	DY-80	12/2	Pre-chilled molybdenum wrought nozzle extension test
14	DD 113, 114	11/13 to 11-18	Hard vacuum start tests
15	E-39-1 to -4	11/5 to 12/10	Lunar survival storage
16	F 3 to 9	11/9 to 12/3	Tank storage and firing test
17	AS-1	12/3	Static acceleration

³Prepared by JPL Technical Section 384.

⁴RMD = Reaction Motors Division of Thiokol Chemical Corporation.

a. Group 12. This group of tests was performed to determine the suitability of the flight-type, stress-relieved wrought molybdenum nozzle material when test-fired after having been pre-chilled to successively lower temperatures. The third and last test (DY-80) was successfully performed on December 2. Previous tests of this group, DY-76 and DY-77, were reported respectively in SPS 37-35, Vol. I, and SPS 37-36, Vol. I.

Following Test DY-77, the burned-out chamber pressure tap was welded closed. The nozzle was pre-chilled with LN₂ to a temperature of -140°F at the chamber end and -260°F at the exit end. The engine was operated, with no degradation of the nozzle, for 30 sec at the maximum thrust level.

This test concluded the pre-chill testing of the molybdenum nozzle. It is concluded, based upon the results of 6 tests (DY-53, 54, 55, 76, 77, and 80)⁵, that the molybdenum nozzle is fully capable of withstanding pre-chill temperatures as low as -200°F with no detriment to its operability in a subsequent firing.

b. Group 14. This group of tests was performed to demonstrate the start characteristics of the engine in a hard-vacuum environment. The test plan specified demonstration firings in a vacuum of about 10⁻⁵ torr with propellants and the thrust chamber assembly (TCA) conditioned to temperatures ranging from 0 to +165°F. The first two tests, DD-111 and DD-112, were reported in SPS 37-36, Vol. I.

Fig. 26 illustrates the thrust chamber installation in the HYPROX facility, including the insulated and temperature-conditioned propellant feed system.

Fig. 27 illustrates the hard-vacuum environmental chamber surrounding the TCA, including the exhaust line leading to the diffusion pump and the exhaust port of the HYPROX system.

The third test (DD-113) was successfully performed on November 13. The start was made in a vacuum of 4.0×10^{-4} torr with propellants conditioned to 0°F.

⁵Tests DY-53, 54, and 55 reported in SPS 37-33, Vol. I, were performed employing an early-design, partially-recrystallized molybdenum nozzle. Tests DY-76, 77, and 80 employed the flight-type, stress-relieved (no recrystallization) wrought molybdenum nozzle material.

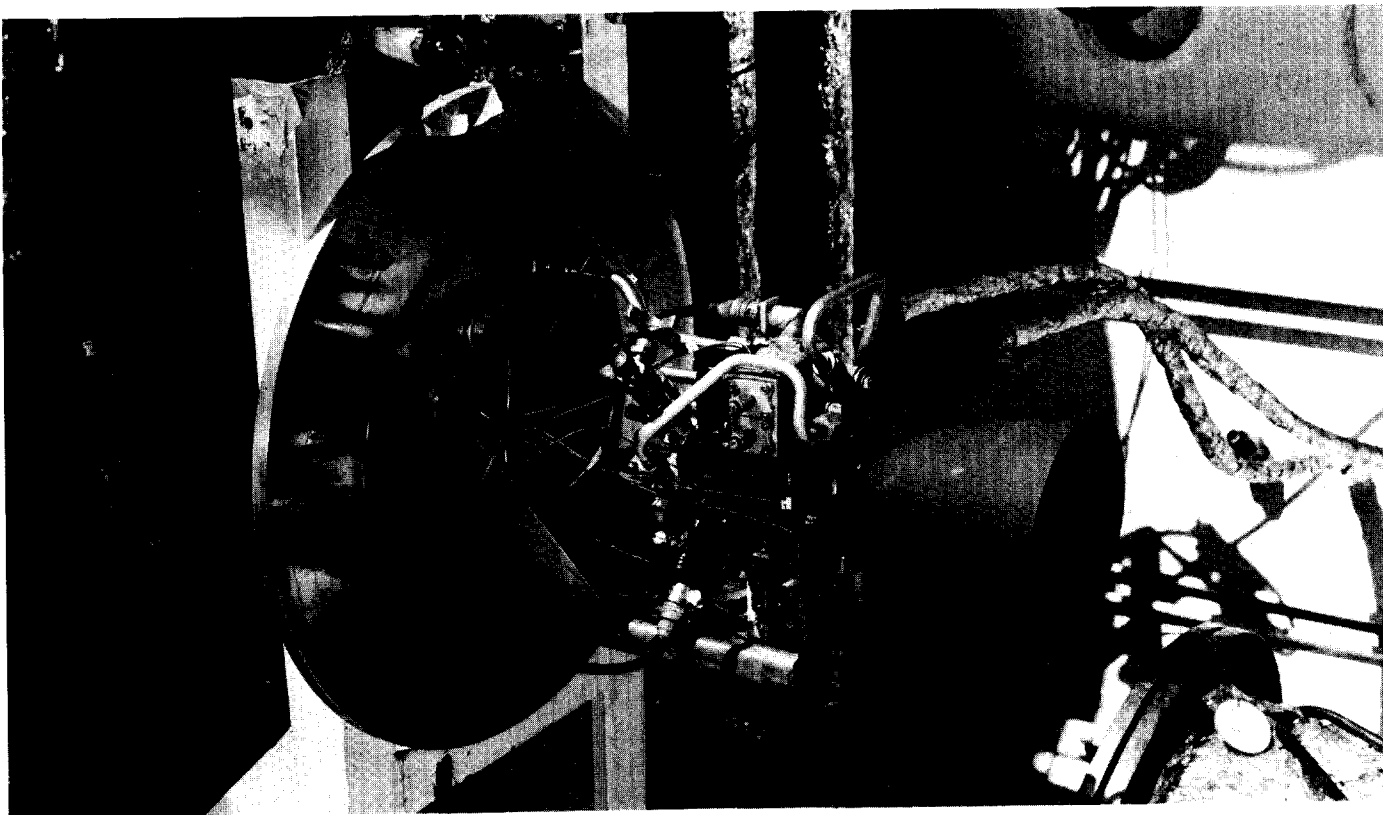


Fig. 26. Thrust chamber installation—hard vacuum (Tests DD 111–114)

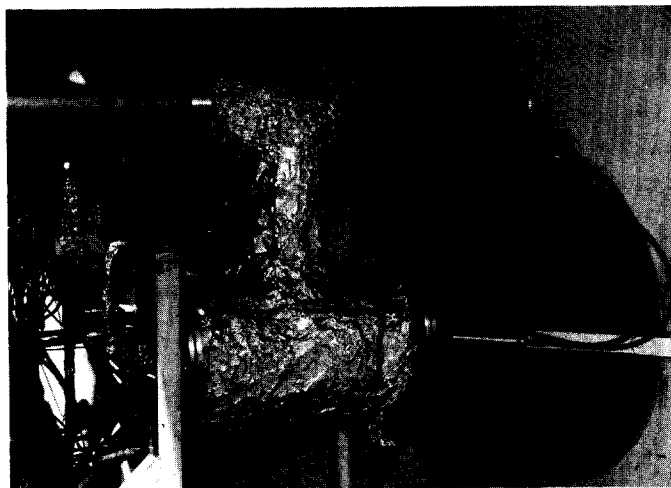


Fig. 27. Hard-vacuum environmental chamber (Tests DD 111–114)

The ignition delay (defined as the interval of time between “shutoff valve starting to open” and initial chamber pressure rise) was about 50 msec. The initial chamber pressure rise was normal, but was followed by a pressure decay and a gradual rise after about 3 sec to the com-

manded mid-thrust level. It is believed that the combination of a local cold spot in the oxidizer circuit (resulting in partial restriction of the flow) and increased fuel viscosity caused the abnormal start transient.

The fourth test (DD-114), with propellants conditioned to $+165^{\circ}\text{F}$, was successfully performed in a vacuum of 4.3×10^{-4} torr on November 18. The start transient was normal in every respect. Ignition delay was comparable to that of the three previous tests made at colder temperatures.

In summary, the average ignition delay for the four tests performed in a hard-vacuum environment (about 4×10^{-4} torr) was 52 msec. In the four tasks, conducted over a temperature range of 0 to $+165^{\circ}\text{F}$, the ignition delays varied only from 47 to 60 msec.

Some measure of the effect of hard vacuum upon the starting characteristics of the thrust chamber can be obtained by comparing the above ignition lag with the average ignition delay experienced in some 32 tests performed at JPL Edwards Test Station with essentially

ambient-temperature propellants under sea-level and 120,000-ft (3.5 torr) simulated altitude conditions. The average ignition delay for this latter group was 30 msec.

It is concluded that on the basis of TCA tests, as well as cold-temperature tests of the S-6 type-approval system, the start characteristics of the TD-339 TCA operating on MON-10 and MMH·H₂O are acceptable as long as propellant temperatures (including the fuel in the jacket) are maintained in the range of 0 to 165°F.⁶

c. Group 15. These tests were performed to demonstrate the ability of the thrust chamber to retain propellants without leakage while exposed to a simulated post-lunar-landing environment. For the purpose of this demonstration the lunar environment and test conditions were defined as follows:

- (1) Temperature range: -300 to +350°F.
- (2) Internal TCA propellant pressure (at upper temperature extreme): up to 1200 psig.
- (3) Duration: 15 days at each temperature.
- (4) Vacuum: not to be simulated.

The storage program to meet these objectives was initiated at JPL Edwards Test Station in November. Two conditioning boxes, capable of cycling between -300 and +300°F, were available and dictated the range of temperature to which the engines could be exposed. Initially, two sea-level thrust chambers were allocated to this storage program.

Each thrust chamber was equipped with two small-volume feed lines (attached to the inlet ports) for containing the fluids. One thrust chamber (SN 509) was to be stored in the conditioning box with oxidizer in the oxidizer side and isopropyl alcohol in the fuel side. The other thrust chamber (SN 511) was to be stored with fuel in the fuel side and Freon-TF in the oxidizer side.

The feed lines for each engine were equipped with suitable fill and vent lines and included a regulated inert nitrogen gas makeup system. Each circuit contained suitable devices to protect the thrust chamber from excessive temperature and pressure.

Following installation in the conditioning boxes, appropriate passages of each thrust chamber (including the

feed lines) were partially filled with propellant or solvent, as appropriate, under a blanket of nitrogen gas. Storage pressures at the high temperature were set in the range of 1000 to 1200 psi. The temperature conditioning profile prescribed for the test involved consecutively: (1) 5 days at high temperature, (2) 15 days at low temperature, and (3) 10 days at high temperature.

The oxidizer thrust chamber was successfully stored (Test E-39-1) under the prescribed conditions from November 5 to December 10. On the third day, a safety diaphragm ruptured in the fuel circuit venting the isopropyl alcohol from the engine. Thereafter, the thrust chamber was stored with only oxidizer under pressure in the oxidizer side of the engine.

During the initial high-temperature and the low-temperature periods (20 days of storage) there was no evidence of leakage or pressure decay in the small volume-locked-up system. Following recycling to +300°F, a small amount of leakage evidenced itself on the second day (22nd day of storage). Pressure started to decay at a rate of about 20 psi/hr in a closed volume of about 10 in.³ Some part of this leakage may be attributed to facility gas check valves which were also exposed to the high temperature. Pressures were maintained at 1200 psi through the nitrogen makeup system. This pressure decay rate subsequently increased to 80 psi/hr by the 10th day (30th day of storage) at which time the experiment was terminated and the thrust chamber removed from the conditioning box.

Subsequent solvent leak checks of the engine showed only one liquid leakage point, and this was at the oxidizer torque motor seal. To date, a complete examination of the thrust chamber and the effect of propellant storage on the various seals has not been performed but will be accomplished in the immediate future.

Lunar survival testing (Test E-39-2) of the fuel thrust chamber was initiated on November 5. On November 7, a rise in pressure in the fuel circuit to about 1250 psi caused a safety diaphragm to rupture, releasing all fuel from the engine. Subsequent examination of the thrust chamber revealed gross leakage through the majority of the seals and a non-operable condition for several critical components (shutoff and throttle valves).

The thrust chamber was replaced with the SN 533 engine and testing (Test E-39-3) was resumed on November 17. Following one day of storage at +300°F, a pressure rise in the fuel side of the thrust chamber presumably

⁶MON-10 refers to mixed oxides of nitrogen containing 10 wt % and 90 wt % N₂O₄.

MMH·H₂O=monomethylhydrazine hydrate.

unseated the shutoff valve poppet and allowed all fuel to vent from the circuit. Subsequent inspection revealed that the nylon seat on the fuel side of the valve had completely disintegrated.

The thrust chamber was replaced with the SN 520 engine, an additional over-temperature protection circuit was added to the temperature conditioning box controls, and testing (Test E-39-4) was again resumed on November 19.

The thrust chamber was successfully stored under 1000-psi pressure at $+300^{\circ}\text{F}$ from November 19 to 23. In a routine check of the system at 0330 hour on November 24, both temperature and pressure were found to be normal. In a subsequent check at 0500 hour on the same day, pressure had decayed to zero and all fuel had leaked from the thrust chamber through the shutoff valve. The sudden loss of fuel did not appear to be the result of box overheating or lack of temperature control.

Examination of the fuel poppet of the shutoff valve again revealed disintegration of the Nylon 101 (Dupont) seat. The failure was manifested by an apparent hardening and embrittlement of the material, discoloration, and the development of cracks and cleavage planes.

Since these tests have revealed an apparent lunar storage survival limitation, additional compatibility tests of this material with fuel are presently planned to determine the limiting conditions of temperature and exposure time to which the nylon can be subjected. The properties of other materials will also be examined to determine whether improved high temperature exposure time could be expected with a substitute material.

d. Group 16. This group of tests was performed to: (1) determine whether a mission profile involving continually decreasing propellant temperature would cause a bubble of helium gas to form at the top of the bladder and be ingested into the engine affecting controllability, and (2) determine whether propellant vapors (particularly oxidizer) in the helium pressurization lines at temperatures below -100°F would condense and freeze and prevent the flow of pressurizing gas to the tanks. Additionally, tests were to be performed duplicating the high-temperature conditions of the S-6 type-approval test environmental program previously performed by the Hughes Aircraft Company in which certain anomalies in thrust response were noted.

A representative feed system was built up on the F-stand at JPL Edwards Test Station by mating a set of bladdered

aluminum tanks supplied by Hughes Aircraft Company with a sea-level engine on a special frame. Fig. 28 shows the two aluminum tanks, the thrust chamber at the right-hand side, and the pressurizing line conditioning unit, all mounted on the base of the temperature conditioning box. The propellant discharge circuits closely duplicated those of the spacecraft in terms of length, diameter and arrangement.

The helium gas pressurizing lines, employing the facility helium supply, were modified to include U-bends about 12 in. from the propellant tank inlets. The U-bends were to serve as a trap for any propellant vapor condensate which might form. Surrounding each U-bend was a conditioning unit containing cooling coils and aluminum powder to transmit the proper amount of chill to the gas lines.

The complete feed system, including the thrust chamber, was enclosed in a conditioning box designed to control temperature between 100 and 30°F and provide a continuous decay rate of about 0.75°F/hr over a 66-hr simulated mission.

The test program was planned to consist of two simulated missions. Each included a midcourse and descent firing and was based upon modified profiles taken from the TCA type-approval test program conducted by RMD and from the S-6 type-approval test program conducted by Hughes Aircraft Company. In both mission profile

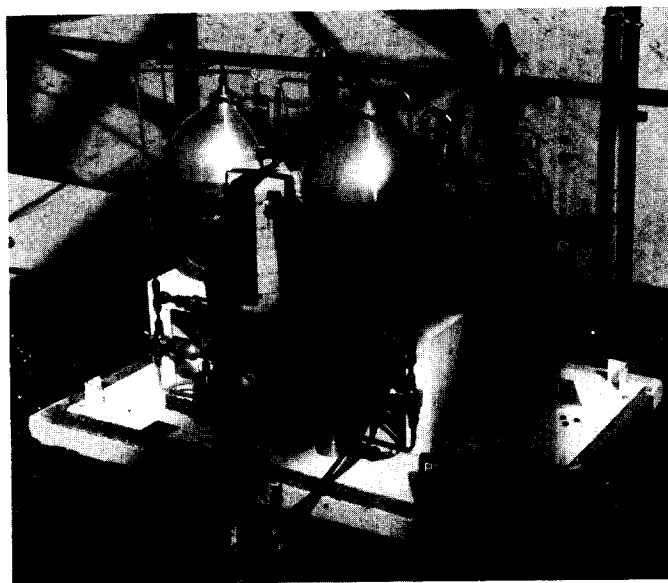


Fig. 28. Feed system and thrust chamber installation (Tests F 3-9)

tests the pressurization lines were to be conditioned to temperatures below -100°F .

The first mission profile test was to be in simulation of the propellant temperature decay which is expected on Leg 2 of the spacecraft. The second mission profile test was to be at a constant temperature of 100°F in simulation of the S-6 type-approval test high-temperature environmental test. Both mission profiles were successfully performed at Edwards Test Station during November and December.

For the first simulated mission the propellants were loaded and conditioned for about 15 hr at 85°F and a pad pressure of 300 psi. The propellant tanks were then slowly cooled in simulation of spacecraft conditions following launch. The pressurization lines were also conditioned at this time. After 15 hr, the tanks were pressurized to full pressure and a simulated midcourse firing conducted (Test F-3).

About 50 hr after this firing a facility helium relief developed a leak. Tank pressure had to be reduced to allow repairs. Previous experience had indicated that a bubble would form inside the bladder as a consequence of the reduction in tank pressure. Accordingly, the mission was terminated, the tanks were reloaded, and the simulated mission was repeated.

Following the repeated midcourse firing (Test F-4), the propellant tanks were cooled further and the helium line was maintained at -128°F . After a total of about 66 hr had elapsed, a simulated lunar descent firing was conducted (Test F-5). At this time the propellant tank temperature was 33°F . The data from the firing gave no evidence of gas ingestion.

The next series of tests (F-6 and F-7) was conducted in nearly the same manner, except that the propellants were maintained at 100°F for the full 66-hr mission. The helium pressurization line was conditioned to -150°F throughout the 66 hr.

Prior to midcourse firing (F-6), a short circuit developed in the shutoff valve circuit. Again, the tanks had to be depressurized to 400 psi to effect repairs. At this pressure level, it was believed gas would not be released from solution, since the system had been under full pressure (720 psi) for only about $\frac{1}{2}$ hr. However, the subsequent simulated descent firing (F-7) results indicated bubble formation in the bladder and gas ingestion during the descent firing.

Since these test results were clouded by the need for depressurization during the simulated mission, the mission profile was repeated. The repeat firings (Tests F-8 and F-9) were performed successfully according to the same constant temperature profile. Conditioning of the helium lines for these last two firings was waived since the propellant vapor freezing information had been successfully obtained on Tests F-6 and F-7.

Examination of the chamber pressure records following Test F-9 showed no evidence of "uncommanded thrust excursions" in the first step to maximum thrust and only a small chamber pressure reduction (about 5 psi) during the second step (5-sec duration) to maximum thrust. It was concluded that there was little or no gas ingestion under constant 100°F propellant temperature conditions and in all likelihood the anomalies noted during the high-temperatures S-6 type-approval test firing were the result of excessive bladder leakage.

The second objective of this group of tests was also successfully accomplished; i.e., to determine if a propellant vapor freezing problem existed in the cold helium pressurizing lines. During the temperature decay mission (Tests F-4 and F-5), the pressurization lines were conditioned to -128°F . During the first constant-temperature mission (Tests F-6 and F-7), the lines were conditioned to -150°F .

In both simulated missions, pressurization was normal and no evidence of blockage was observed. Since the lowest temperature observed in these lines was about -90°F during solar-thermal-vacuum testing of Spacecraft SN 1, it is concluded that current thermal control of these lines is acceptable and that pressurization line obstruction due to oxidizer vapor freezing will not occur.

e. Group 17. A supplemental type-approval test program was initiated at Edwards Test Station with the spare RMD type-approval test engine (SN 535) to subject the engine sequentially to several nonfiring mission environments followed by several firings under simulated mission firing conditions. This program is intended to add to the recently-completed TCA type-approval test program at RMD by exploring the effects of those environments not previously tested at RMD.

The first test in this new group involved exposing the TCA to the non-firing environment of static acceleration during a simulated launch. The engine was mounted horizontally on the rotating arm of a centrifuge unit, pressurized with solvents to 300-psi pad pressure, and

subjected to 4 min of 12-g followed by 8 min of 5-g static acceleration along its longitudinal axis. Solvent flow tests, before and after static acceleration, indicated no change in the throttle valve calibration or the operability of the shutoff valve.

Other tests in this group to be accomplished in the immediate future include exposure to launch and landing shock, mission duration thermal-vacuum storage and firings, thrust vector alignment measurements, margin limit firings, and TCA temperature gradient firings.

6. Flight Systems Support⁷

In order to assure that the thrust chamber assemblies assigned to the flight program have not undergone any degradation during the various phases of pre-flight testing of the spacecraft, a set of procedures has been generated for checkout of each TCA prior to its installation on the spacecraft.

To accomplish the TCA checkout a set of ground support equipment was designed and fabricated by JPL with Thiokol-RMD assistance. The equipment includes: (1) a vacuum and leak check assembly, (2) a solvent flow bench, and (3) control electronics and instrumentation. This equipment has been installed at the Air-Force Eastern Test Range (AFETR) and is in process of checkout in anticipation of the receipt of the first flight spacecraft.

Upon receipt of the TCA's at AFETR, they will be solvent-flow-checked to establish their steady-state and dynamic characteristics. Information obtained will be compared to that obtained when the TCA was originally acceptance-tested by the contractor.

This effort should establish a high degree of confidence in the integrity of the thrust chamber assemblies and provide a useful tool in determining any anomalies of this subsystem as well as a basis for trouble shooting.

M. Spacecraft Vehicle

1. Spacecraft Vehicle

Vehicle activities reported in this section include the design and fabrication of spaceframe and substructures of the assembly of the complete vehicle for flight and test spacecraft.

a. Vehicle status. The S-15 Drop Test Vehicle (A-21) is being upgraded for three dimensional landing stability tests. It has been equipped with new landing gear position potentiometers and side load strain gage assemblies. Otherwise, it is nearly identical in appearance to the T-1 vehicle from which it was made (Fig. 29).

The detailed layout and design engineering on the SC-5 test vehicle (A-21A) continues. A directive to remove Camera 4 has been issued.

The general arrangement drawing and engineering planning specification for the S6A test vehicle (A-21A) has been released.

The engineering planning specifications, detail and assembly drawings of the spaceframe and detail drawings for the landing gear were released for the S-11 test vehicle (A-21A), and the engineering planning specification for the S-12 test model (A-21A) was to be released the latter part of December 1965.

The design of the S-14 and S-16 vehicles will be discussed in the next SPS and continued in future issues.

b. Touchdown dynamics experiment A-21A. This experiment involves the development of a lunar contact switch for mounting to the base of each landing leg foot pad and crushable block. The contact switch consists of a sensitive contact element, a mounting base, and electrical

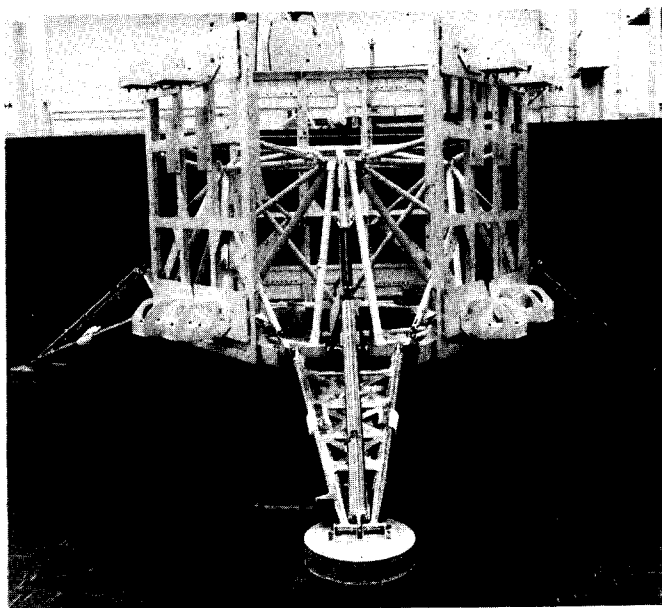


Fig. 29. S-15 test vehicle

⁷Prepared by JPL Technical Section 384.

leads for connecting to the touchdown dynamics experiment harness. One of these switches is to be attached to each of the foot pads and crushable blocks by means of screws and plate nuts, to allow replacement in case of damage. The sensitive element contains a plated gold circuit that is opened when the element is broken by physical lunar contact. This design is being further developed to determine the optimum sensitive element dimensions to withstand the operational requirements and the environmental requirements.

2. Mechanisms

a. Omnidirectional antenna mechanisms. The two omnidirectional antenna mechanisms (A and B) use tapered fiberglass cloth tubes bonded to aluminum hinge bracket assemblies. They deploy and lock the two omnidirectional antennas into operating position before separating from the *Centaur*. Omnidirectional antenna mechanisms A and B for SC-3 have been delivered to bonded stores. The mechanisms for SC-1, SC-2, T-21, and SP-1 had been previously delivered to bonded stores.

b. Antenna/solar panel positioner. The antenna/solar panel positioner provides mounting, positioning, position indicators, and launch and landing locks for the planar array antenna and solar panel. The antenna support structure includes controllable axes for roll, elevation, solar, and polar rotations. The drive for each axis uses a solenoid stepper motor for power and a wire-wound potentiometer for position indication.

c. Separation sensing and arming devices. Three separation sensing and arming devices per spacecraft provide redundant interruption for the postseparation squib firing circuits and vernier engine solenoid circuit until after the spacecraft separates from the *Centaur*. A tolerance review of the switch actuators disclosed marginal actuation conditions under worst case tolerance buildups. The switch actuators have been redesigned to assure positive operating margins. In addition, a third switch has been incorporated into each device to provide triple redundancy. New devices of the improved design have been built, tested, and delivered for SC-1 and SC-2.

d. Pyrotechnic devices. Nine of the thirteen squib-actuated pin pullers on the spacecraft have been modified by the addition of insulation from chassis ground and of resistors for static bleed protection. A request for proposals for squibs that do not short to ground after firing has been issued to qualified vendors. It is expected that nonshorting squibs will be available for spacecraft use by May 1966.

e. Soil mechanics/surface sampler. The soil mechanics/surface sampler instrument is a variable extension-retraction device providing the capability of controlled motion and adjustment in azimuth, elevation, and radial planes. The mission objective of the soil mechanics/surface sampler experiment subsystem is to relay to Earth data regarding the physical and mechanical property characteristics of the lunar surface. All required scientific data will be obtained by manipulation of the surface sampler during its selected task command mode. Potentiometer, accelerometer, and strain-gage instrumentation on the surface sampler will measure and record induced motion, acceleration, load levels of the instrument during lunar excursion, contact, and excavation.

f. Alpha scattering instrument deployment mechanism. The vibration and shock levels for the deployment mechanism have been defined as 25-g shock maximum—12-g vibration longitudinal, and 10-g vibration lateral maximum. These values are within the limits for which the deployment mechanism was designed.

The design of the mechanism was completed and all the detail drawings (approximately 60) have been prepared. Two of six assembly drawings have been updated and checked. The remaining drawings were to be completed by approximately December 15, 1965.

A test fixture to simulate operation under lunar-gravity conditions has been designed. Detailed design of the unit was to have been completed by December 22.

A layout of a reduced height model was prepared as a possible approach toward obtaining a reduction in weight of the alpha scattering experiment. The purpose of reducing the height was to permit the deployment mechanism and instrument to be moved inboard under TV Camera 3. This might have resulted in saving approximately 0.20 lb on the deployment mechanism and 0.75 lb of supporting structure. However, because of increased thermal problems, this approach was abandoned.

Stress analysis of major mechanism components was completed. The thermal aspects of the design have been reviewed and the thermal control surface treatments established for the unit.

A foam rubber for use as a bumper material is being tested. It is being cleaned at 360°F for 10 days and will then be tested at 260°F to see if any contaminants remain. Stretch tests over a period of 3 wk were conducted on both nylon and dacron braided line. From these tests, it

appears desirable to prestretch the line prior to installation. It is now undergoing thermal-vacuum, ultraviolet, humidity, and strength tests.

g. Fabrication. A complete set of detail drawings has been released for the fabrication of an engineering model. Pin puller requirements have been defined and negator springs and ball bearings for the lunar simulator have been ordered. The new contour cable wire has been fabricated and the first contour cable harness is nearing completion.

N. Space Flight Operations

1. Surveyor Pulse Code Modulation (PCM) Telemetry Bit Error Rates

A quick-look report of Command and Data Handling Console/Deep Space Instrumentation Facility (CDC/DSIF) compatibility tests conducted at the remote DSIF Stations 11 (Goldstone), 42 (Canberra), and 51 (Johannesburg) was recently issued (Ref. 8). These tests were conducted according to the CDC/DSIF compatibility test procedures (Ref. 9) primarily to assure the adequacy of CDC equipment and DSIF station equipment to conduct the *Surveyor* mission.

a. Bit Error Rate Test Results. Bit error rate probabilities versus unmodulated signal-to-noise ratios in the bandpass of the CDC channel selector for the various *Surveyor* bit rates are given in Fig. 30(a) for DSIF 11 (Goldstone), Fig. 30(b) for DSIF 42 (Canberra), and Fig. 30(c) for DSIF 51 (Johannesburg). In general, the data scatter from DSIF-11 (Goldstone) is much worse than the other two stations. Previous short-loop bit error rate tests 30(a) at DSIF 11 with the DSIF station receiver out of the signal loop, have produced data plots with far less scatter than that shown in Fig. 30(a). For all stations it is felt that the PCM data restoration circuits are performing in a less than optimum manner. Corrective action on the system seems mandatory, both to reduce data scatter and to move the mean bit error rate curves to the left.

b. Frequency multiplexed telemetry. Two series of tests were conducted with frequency multiplexed telemetry. One was in the FM-FM mode with PCM data at

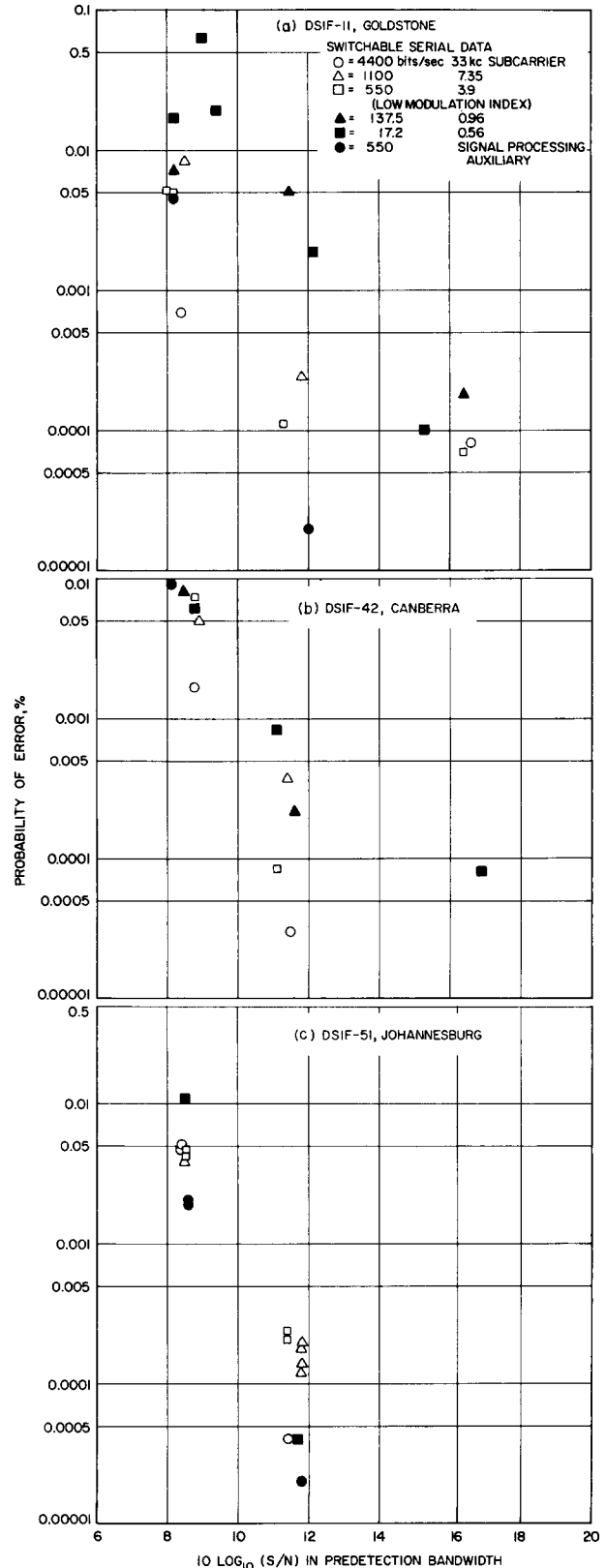


Fig. 30. Probability of error vs signal-to-noise ratio in predetection bandwidth

4400 bps (33-kc subcarrier oscillator) mixed with the six analog channels with subcarrier oscillator center frequencies of 52.5, 70, 90, 110, 130, and 150 kc. The other series of tests were conducted in the FM-PM mode with PCM data at 1100 bps (7.35-kc subcarrier oscillator) mixed with the three strain gage channels with subcarrier oscillator center frequencies of 1.7, 3.0, and 5.4 kc. Data obtained from these tests are summarized in Tables 12 and 13. Bit error rates obtained are shown in Table 14.

The multiplexed FM-FM data show that the signal-to-noise ratios at the outputs of the 33 and 52.5-kc subcarrier oscillators are adequate for proper demodulator performance at threshold and consistent among stations. However, the signal-to-noise improvement factors are below predicted values based on the standard wideband improvement formula. One apparent discrepancy is the performance of the 33-kc discriminator. In these multiplex telemetry tests, the signal-to-noise ratios were degraded through this discriminator by an average of 8 db. In the single channel PCM-FM-PM case (with the 33-kc

subcarrier oscillator), the signal-to-noise ratios were improved. The reason for this difference has not yet been determined. The multiplexed FM-PM data are also consistent among the stations, and in this case, the data from the 1100 bps channel (7.35-kc subcarrier oscillator) agree with that obtained during the telemetry error rates test of the 7.35-kc channel in the PCM-FM-PM mode. Again, the improvements are derived from the wideband improvement formula and the bit error rate obtained is close to that expected.

O. Aerospace Ground Equipment

1. Ground Support Equipment (GSE)

The television line selector unit in the television test rack has been redesigned to add emergency mode capability. In addition, changes were made to increase effec-

Table 12. Multiplexed telemetry data, FM-FM mode (4400 bps plus six analog accelerometer channels)^a

Station	Test	Demodulator in	Demodulator out	Improvement	33-kc bandpass	33-kc subcarrier output	Improvement	52.5-kc bandpass	52.5-kc subcarrier output	Improvement
DSIF-11	11-7 ^b	13.3	30.3	17.0	37.7	27.5	-10.2	35.5	37.4	1.9
	11-12 ^b	10.9	27.0	16.1	38.4	27.8	-9.6	35.7	36.6	0.9
DSIF-51	11-7	10.4	26.1	15.7	36.5	33.4	-3.1	35.9	36.8	0.9
	11-12	10.6	27.1	16.5	36.4	33.2	-3.2	35.2	34.2	-1.0
DSIF-42	11-7	9.7	24.9	15.2	36.0	28.1	-7.9	31.8	35.9	4.1
	11-12	10.5	26.0	15.5	35.6	25.3	-10.3	34.6	33.8	-0.8

^aSignal-to-noise ratios measured in db.
^bRef. 9.

Table 13. Multiplexed telemetry, FM-PM mode (1100 bps plus three analog strain gage channels)^a

Station	Test	7.35-kc bandpass	7.35-kc subcarrier output	Improvement	5.4-kc bandpass	5.4-kc subcarrier output	Improvement
DSIF-11	11-7 ^b	7.5	8.0	0.5	9.2	6.5	-2.7
	11-12 ^b	7.6	7.4	-0.1	8.2	6.8	-1.4
DSIF-5	11-7	7.9	10.2	2.3	6.2	4.0	-2.2
	11-12	7.4	7.5	0.1	7.8	5.8	-2.0
DSIF-42	11-7	7.9	8.8	0.9	7.3	7.3	0
	11-12	7.9	8.7	0.8	8.4	6.9	-1.5

^aSignal-to-noise ratios measured in db.
^bRef. 9.

Table 14. Bit error rates multiplexed telemetry

Signal input	DSIF-11		DSIF-51		DSIF-42	
	Signal-to-noise input	Bit error rate	Signal-to-noise input	Bit error rate	Signal-to-noise input	Bit error rate
33 kc plus six analog accelerometer channels FM-FM	20.2	0	—	—	20.1	0
	13.3	0	13.2	0	13.2	0
	10.9	0.03×10^{-3}	10.4	0	9.7	0
	7.5	0	7.1	0	7.0	0
7.35 kc plus three analog strain gage channels FM-PM	14.3	0	15.2	0	16.0	0
	11.8	0	11.7	0.29×10^{-3}	11.5	0.21×10^{-3}
	7.5	5.9×10^{-3}	7.9	11.2×10^{-3}	7.9	8×10^{-3}

tiveness in the normal mode. A television light shroud, required for star mode television testing, has been designed, fabricated, and tested. It is designed to work with target illumination levels of 0.02 ft.-L.

2. STEA 3 Trailer

a. Installation. A substantial portion of the STEA 3 equipment has been installed in a 40-ft trailer for transportation and use with T-21 DSIF compatibility tests at Goldstone and drop tests at El Segundo. DSIF compatibility testing was completed at Goldstone on October 11, and the trailer and STEA equipment were then moved

to El Segundo. Preparations are now under way for use of this equipment in T-21 drop testing on an inclined simulated lunar surface (Fig. 31) scheduled for the near future.

b. Power equipment. Power equipment required for operation of the STEA has been installed, and the remaining required STEA checkout equipment has been interconnected with the T-21 drop test area and the STEA power sources. Installation, interconnecting, and preparation of STEA 3 equipment is 90% complete and is scheduled for 100% completion prior to the T-21 drop test.

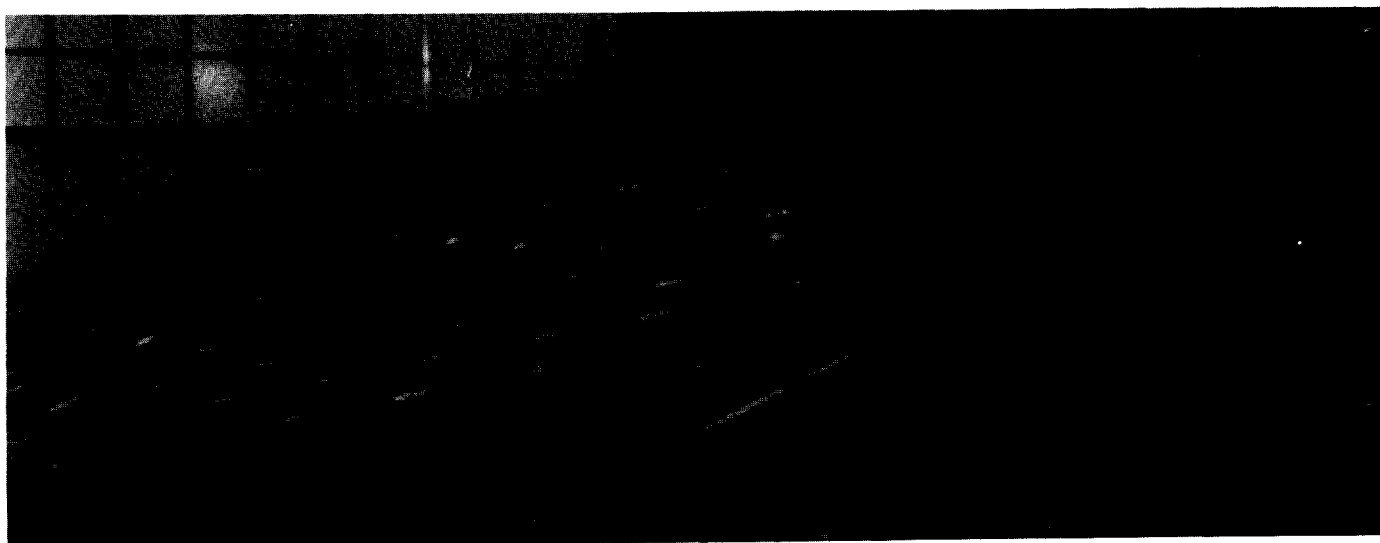


Fig. 31. Inclined simulated lunar surface

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